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APPLICATION STUDY OF FILAMENTARY COMPOSITES IN A COMMERCIAL JET AIRCRAFT FUSELAGE

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FOREWORD

This is the final report for NASA Contract NAS1-11162, "Application Study of Filamentary Composites in a Commercial Jet Aircraft Fuselage." The program was accomplished during the time period from October 1971 to June 1972 and was monitored by Mr. H. Benson Dexter of the Composites Section, Materials Application Branch, NASA Langley Research Center.

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ABSTRACT

A study of applications of filamentary composite materials to aircraft fuselage structure was performed. General design criteria were established and material studies conducted using the 727-200 forebody as the primary structural component. Three design approaches to the use of composites were investigated: concept 1, uniaxial reinforcement of metal structure; concept 2, uniaxial and biaxial reinforcement of metal structure; and concept 3, an all-composite design. Materials application studies for all three concepts were conducted on fuselage shell panels, keel beam, floor beams, floor panels, body frames, fail-safe straps, and window frames. Cost benefit studies were conducted and developmental program costs estimated, also for all three concepts. On the basis of weight savings, cost effectiveness, developmental program costs, and potential for early application on commercial aircraft, the concept 1 design is recommended for a 5-year flight service evaluation program.

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APPLICATION STUDY OF FILAMENTARY COMPOSITES IN A COMMERCIAL JET AIRCRAFT FUSELAGE

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1.0 SUMMARY

A study of applications of filamentary composite materials to aircraft primary fuselage structures was performed. The Boeing 747, 737, 727-200, and the 707-320 aircraft were reviewed to determine the component best suited for the study, and a 4.57-m (180-in.) long section of the 727-200 forebody was chosen as the demonstration component. The structural components studied are shown in figure 1. The study established a three-level approach to the use of composites, namely: concept 1, uniaxial reinforcement of metal structures; concept 2, uniaxial and biaxial reinforcement of metal structures; and concept 3, an all-composite design.

General design criteria were established for all three concepts, and material selection studies were conducted to determine the most suitable composite material for each design concept. High-strength graphite epoxy was chosen for all concepts because of its lower cost. The adverse effect of residual tensile thermal stresses induced in the metal-reinforced components would be reduced by applying thermal stress alleviation techniques during bonding. The weight savings achieved in this phase of the study are shown in table 1.

TABLE 1.—SUMMARY OF STUDY SECTION WEIGHT SAVINGS

Component	Baseline		Concept 1		Concept 2		Concept 3	
	kg	lbm	kg	lbm	kg	lbm	kg	lbm
Shell								
Upper quadrant	144	317	115	254	112	248	111	245
Side quadrant	211	465	171	377	144	318	137	302
Lower quadrant	168	371	143	316	141	310	152	335
Floor beams	57	126	39	86	38	84	38	84
Floor panels	53	116	31	69	31	69	31	69
Keel beam	22	48	14	30	14	31	(a)	(a)
Total weight	655	1443	513	1132	480	1060	469	1035
Weight saving	–	–	141	311	175	383	186	408
% weight saving	–	–	21.5	21.5	26.5	26.5	28.2	28.2
Weight of composite	–	–	34.4	75.8	60.6	133.6	166.3	366.6
CEF[b]	–		4.10		2.87		1.11	

[a]Included in lower quadrant panel
[b]Composite efficiency factor

Cost benefit studies were conducted for all three concepts. The weight savings achieved in the study section were extrapolated to determine savings for a total fuselage, and fabrication costs were estimated for a 300-airplane production program. The results of the economic analysis indicated that the concept 1 design was cost effective for a graphite

composite price of $132/kg ($60/lbm) and the concept 3 all-composite design was cost effective at a composite price of $77/kg ($35/lbm). The results of this analysis indicated that concept 2 would not become cost effective until the composite price is less than $44/kg ($20/lbm). The effect of a reduction in the material cost in this concept was not sufficient to offset the estimated increased production costs.

Detailed developmental program plans were established, and these are presented for all three concepts. The developmental program costs were estimated and the relative costs are defined as shown.

Concept	Relative Developmental Program Costs
1	1.0
2	2.4
3	3.3

The developmental program schedule for each concept is summarized in figure 2.

On the basis of weight savings, cost effectiveness, developmental program costs, and potential for early application on commercial aircraft, the concept 1 design is recommended for a 5-year flight service evaluation program.

2.0 INTRODUCTION

Studies conducted on composite-reinforced metal structures and all-composite structures indicate that the use of these new materials and concepts will significantly reduce aircraft structural weight. To apply these new concepts to full-scale primary aircraft structure for commercial service will require an extensive developmental program. Before committing funds for this program, however, the cost benefits to commercial airlines must be established. To properly assess the ability of these new materials to maintain structural integrity for the life of an aircraft, the developmental program must consist of design and analysis studies, material selection and process evaluation studies, fabrication procedure studies, ground testing, and flight service evaluations.

The Boeing Company 747, 737, 727-200, and 707-320 aircraft were reviewed as candidate components on which to conduct the applications study. The 727-200 forebody was chosen as the primary structural component for the study because it contained major production splices and the loads and skin gages are representative for an aircraft of this size. The applications study was conducted for three design concepts: concept 1, uniaxial reinforcement of metal structures; concept 2, uniaxial and biaxial reinforcement of metal structures; and concept 3, an all-composite design. The primary structural components within this body section that were evaluated for composite application are the following:

- Fuselage shell panels
- Keel beam
- Floor beams
- Floor panels
- Fuselage frames
- Window frames
- Fail-safe straps

General design criteria were established for all three concepts based on existing 727-200 criteria and previous test programs. Material selection studies were conducted to determine the composite best suited for each design concept. The general design criteria and material selection study are discussed in section 3.0.

During the study, the structural components were designed and analyzed for each of the three concepts. Weight savings analyses and cost benefit studies were performed for each concept. Developmental program cost estimates, including a 5-year flight service evaluation period, were obtained for all concepts. The concept studies, weight savings, and cost benefit studies are also discussed in section 3.0. Section 3.6 defines the relative developmental program costs and section 3.7 describes the concept recommended for flight service evaluation.

The demonstration program plan for all concepts consists of five phases:

- Phase I—developmental program
- Phase II—design, analysis, and engineering verification tests
- Phase III—fabrication and quality assurance
- Phase IV—full-scale ground tests and documentation
- Phase V—flight service evaluation

A general discussion of the developmental program is contained in section 4.0 for all three concepts. Detailed test programs, manufacturing procedures, and quality assurance procedures for concepts 1, 2, and 3 are contained in appendixes A, B, and C, respectively.

U.S. customary units were used for calculations throughout the study. These units were converted to SI units as adopted by the Eleventh General Conference on Weights and Measures, Paris, October 1960. Conversion factors for SI to U.S. customary units are presented in appendix D. A table of standard densities used in this study is presented in appendix E.

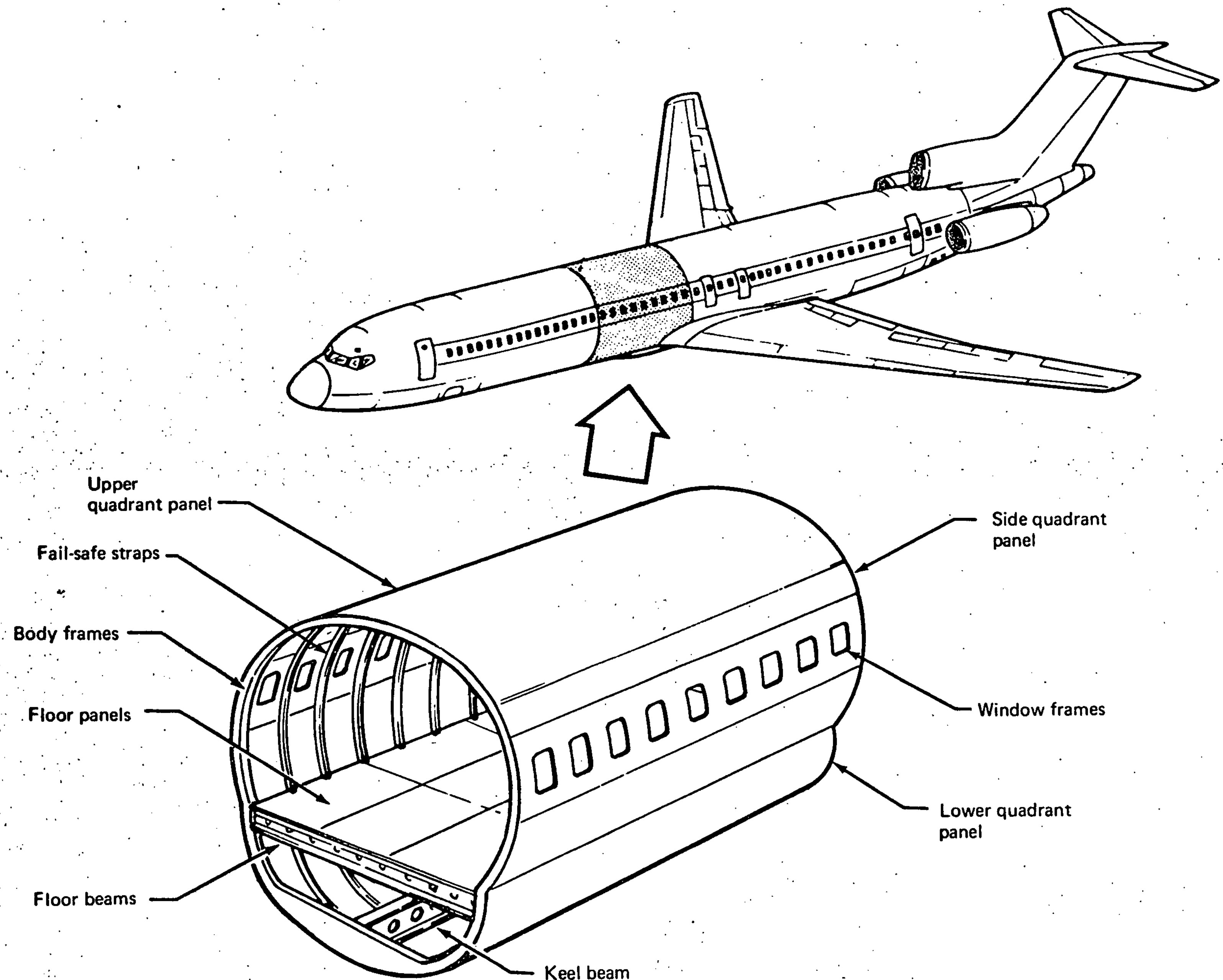
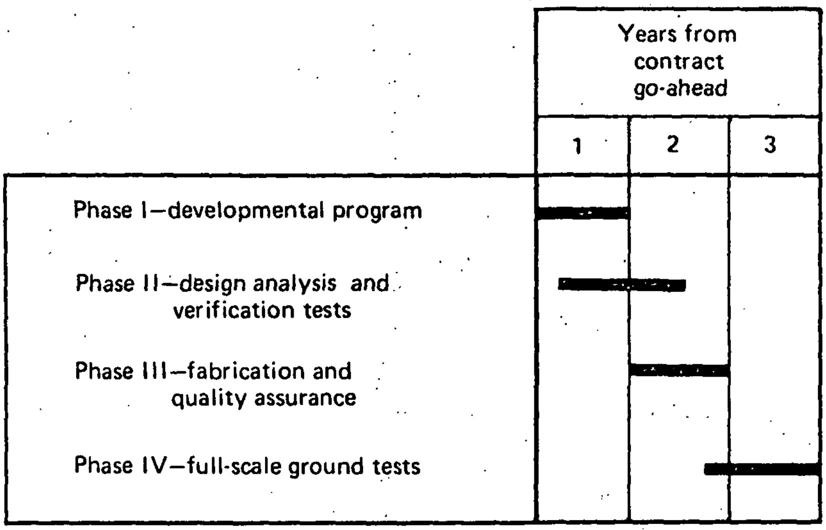
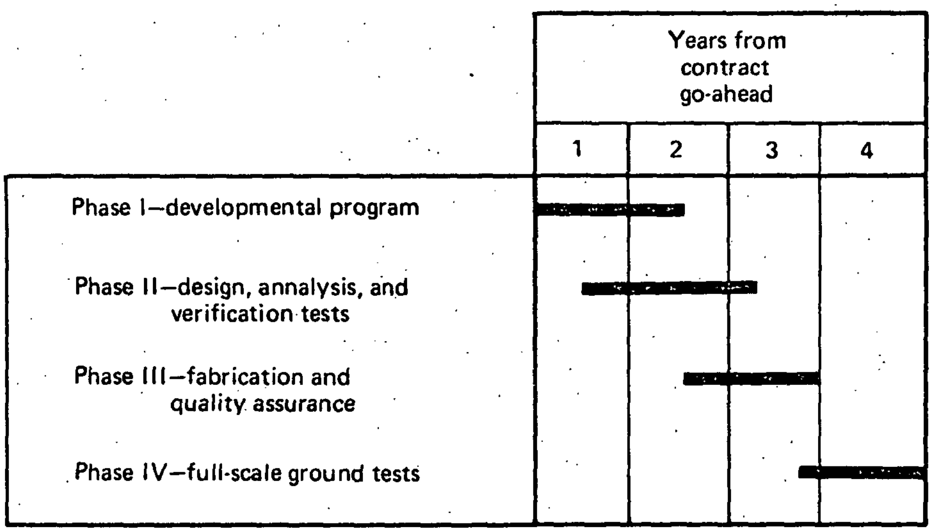


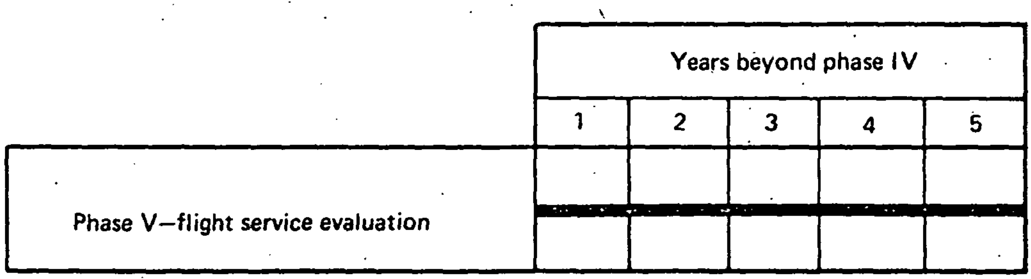
FIGURE 1. —STRUCTURAL COMPONENTS STUDIED



CONCEPT 1—PHASES I–IV



CONCEPT 2 AND 3—PHASE I–IV



ALL CONCEPTS—PHASE V

FIGURE 2.—DEVELOPMENTAL PROGRAM SCHEDULES

3.0 APPLICATION ANALYSIS

3.1 AIRCRAFT AND COMPONENT REVIEW

All of the current Boeing airplane models were initially considered as potential flight test aircraft for this study. The 747 aircraft was dismissed as a candidate when preliminary cost estimates indicated that the developmental program funding level was out of reasonable range. The 707-320 fuselage was dismissed because major cutouts exist in the fuselage sections close to the wing intersection where the skin panels are designed for other than minimum gage considerations. The 707-320 was also dismissed due to uncertainty regarding the future production schedule. The 737 aircraft was dismissed since almost the entire fuselage contains minimum-pressure-designed gages.

The 727-200 aircraft was selected as the best candidate aircraft on which to conduct the applications study. The production schedule carries far enough into the future for possible installation of the test structure in the production line. The test section in the forebody is the stretch portion, thus allowing relatively easy installation at the major body splices. This section also contains no major cutouts, which will reduce the developmental program costs associated with complex structural reinforcement around large cutouts. The fuselage section selected for study is shown in figure 3.

The structural elements considered for composite application were:

- Fuselage frames
- Window frames
- Fail-safe straps
- Floor beams
- Floor panels
- Keel beam
- Tension skin panels (upper quadrant)
- Compression skin panels (lower quadrant)
- Shear skin panels (side quadrant)

Initial studies indicated that composites could not be applied to fuselage frames in a cost-effective manner. The composite-reinforced concept resulted in minimum metal gages that increased the manufacturing costs. The engineering development costs associated with all-composite fuselage frames were considered too high for the small total weight saved. Composite-reinforced and all-composite window frames were also considered unfeasible due to the costly manufacturing processes required to gain the small potential weight saving. The

design of fail-safe straps is influenced by the overall skin panel design; therefore, fail-safe straps were not studied as separate structural components. The remainder of the components listed were designed and analyzed for each concept.

3.2 GENERAL DESIGN CRITERIA

General design criteria were established for the applications study to ensure that the flight test vehicle would equal or exceed the existing 727-200 in ultimate strength, fatigue resistance, and damage containment.

The reinforced components were designed to a requirement that the metal by itself would carry limit load, and composite reinforcing would be added to obtain ultimate loads. Since ultimate load is 1.5 times limit load, this resulted in a metal-to-equivalent-composite ratio of 2 to 1. Initial designs of the fuselage skin panels were not constrained to this criterion. When the skin panels were redesigned to this criterion, a reduction in the weight savings of approximately 10% resulted. The criterion provides adequate safety and design confidence for the flight test vehicle.

The fuselage skin and stringer material in the uniaxial reinforced aluminum construction was designed to the same longitudinal stress level as the existing structure, and the allowable skin hoop pressure stresses were determined from equivalent damage-containment parameters. The effect of residual thermal stresses on the operating fatigue cycles did not cause a reduction in the ultimate design stresses, since the skin panels in the selected section are not fatigue critical for bending stresses, and the thermal stresses would be controlled during fabrication by applying thermal stress alleviation techniques.

An ultimate strain level of 0.15 mm/mm (0.006 in./in.) was imposed on the reinforced titanium and all-composite designs. This criterion was developed on the basis of the results of the NAS1-8858 studies (see ref. 1). The allowable pressure stresses for the reinforced titanium and all-composite designs were determined from equivalent damage-containment parameters.

A criterion was established that all layers of composite at splice boundaries would terminate on bonded stepped load transfer fittings. This was established from NAS1-8858 fatigue and ultimate test results, as well as other Boeing research which indicated that this configuration is the most structurally efficient.

A design and fabrication criterion was established that the composite would be cured and bonded to the stepped load transfer fitting as a subassembly. This reduces the manufacturing complexity and increases the quality assurance of the composite components. Design criteria were also established to ensure that, wherever possible, composite components would be exposed in the final assembly to allow in-place inspection of the components during flight service evaluation.

3.3 MATERIAL STUDIES AND SELECTION

The composite reinforcing materials initially considered for the applications study were boron/epoxy, graphite/epoxy, and PRD-49. Fiberglass was not considered as a primary reinforcing material.

Boron/epoxy was considered for all reinforcing applications and for the all-composite components, because the thermal expansion coefficient of boron, compared to that of aluminum and titanium, makes it more compatible than graphite. However, because of the higher material and machining costs, boron was eliminated from the study.

High-strength graphite/epoxy composite was selected as the major reinforcing and all-composite material. The initial design studies were conducted using graphite layers 0.177 mm (0.007 in.) thick. However, because a considerable weight savings was realized in minimum-gage areas when the graphite layer thickness was changed to 0.127 mm (0.005 in.), the final designs incorporated the thinner graphite layers.

PRD-49 was not considered as a primary reinforcing or all-composite material because of its relatively low compression properties. However, this material was incorporated into a floor beam concept for comparison to a graphite design.

The three graphite/epoxy systems selected for test and evaluation in the developmental program, together with their typical properties, are shown in table 2. The Hyfil material was selected on the basis of some preliminary Boeing IR&D test data that showed good repeatability of tensile strength and modulus properties. In addition, it can be obtained in continuous lengths in widths up to 406 mm (16 in.). This width reduces the fabrication costs involved in laying down large areas of composite for the concept 2 and 3 designs. The Narmco and Hercules systems were chosen as representative materials from U.S. suppliers. All graphite material will be 0.127 mm (0.005 in.) thick.

TABLE 2.—TYPICAL MECHANICAL PROPERTIES[a] FOR CANDIDATE COMPOSITES

Fiber	Resin system	Fiber				Composite						
		Tension modulus		Tension strength		Volume	Tension modulus		Flexural strength		Interlaminar shear	
		GN/sq m	lbf/sq in.	MN/sq m	lbf/sq in.	fraction	GN/sq m	lbf/sq in.	MN/sq in.	lbf/sq in.	MN/sq m	lbf/sq in.
Hyfil 2711	828 DDS	193.06	28×10^6	2413.25	350 000	0.67	131.00	19×10^6	1689.27	245 000	89.64	13 000
Modmor III	Narmco 5209	227.53	33×10^6	2413.25	350 000	0.61	124.11	18×10^6	1379.00	200 000	96.53	14 000
Hercules AS	3M PR288	193.06	28×10^6	2413.25	350 000	0.60	117.22	17×10^6	1379.00	200 000	110.32	16 000

[a]Manufacturers' data

The adhesive systems selected for evaluation in the developmental program are shown in table 3. Test data obtained under Boeing IR&D studies indicate that the environmental stability of the 450° K (350° F) curing adhesives is far superior to that of the 394° K (250° F) curing adhesives. On the basis of these tests, the AF 30 system is presently being considered as the primary load transfer bond material between the composite and step fittings in all concepts. The 394° K (250° F) curing system is presently being considered for bonding the composite and step end fittings to the metal structure in the concept 1 and 2 designs to minimize the residual thermal stresses. The two 394° K (250° F) curing systems will be evaluated during the developmental program, and the final choice will be based on environmental stability characteristics.

TABLE 3.—CANDIDATE ADHESIVE SYSTEMS

Adhesive	Description
AF 30	450°K (350°F) curing nitrile phenolic
Hysol EA 9628	394°K (250°F) curing modified epoxy
3M powder adhesive	394°K (250°F) curing modified epoxy

The material selected for the reinforced aluminum concept consisted of conventional alloys of 2024-T3 and 7075-T6. The 6Al-4V alloy in the annealed condition was selected for the reinforced titanium and all-composite concepts. Design properties for these metals are shown in table 4.

TABLE 4.—CANDIDATE METAL DESIGN PROPERTIES

Property		Aluminum clad 2024-T3	Aluminum bare 7075-T6	Titanium Ti-6Al-4V ann.
Tension ultimate	MN/sq m	413.7	524.0	923.9
	lbf/sq in.	60 000	76 000	134 000
Tension yield	MN/sq m	310.3	455.1	868.8
	lbf/sq in.	45 000	66 000	126 000
Compression yield	MN/sq m	255.1	462.0	910.1
	lbf/sq in.	37 000	67 000	132 000
Shear ultimate	MN/sq m	262.0	317.2	544.7
	lbf/sq in.	38 000	46 000	79 000
Tension modulus	GN/sq m	72.4	71.0	110.3
	lbf/sq in.	10.5×10^6	10.3×10^6	16.0×10^6
Compression modulus	GN/sq m	73.8	72.4	113.1
	lbf/sq in.	10.7×10^6	10.5×10^6	16.4×10^6
Shear modulus	GN/sq m	27.6	26.9	42.7
	lbf/sq in.	4.0×10^6	3.9×10^6	6.2×10^6

3.4 CONCEPT STUDIES

The following concept studies were developed with certain constraints imposed on the designs. These design constraints are defined as follows:

- The outer surface of the fuselage study section was kept the same as the existing section.

- The distance between the outer fuselage surface and the surface defined by the inner flange of the body frames and the spacing of the body frames were also the same as the existing structure. These design constraints allowed for installation of existing interior panels and maintained existing passenger seat spacing.

- The stringer spacing used in all design concepts was kept identical to the existing structure.

3.4.1 Floor Beams

The floor beam is a single-span beam loaded at four locations by the passenger seat tracks. The beam is designed to withstand a forward crash condition of nine times gravitational force. The loading imposes an upward load on one beam and a downward load on the next. The web of the beam contains a series of cutouts that accommodate control cables and hydraulic, electrical, and air conditioning services. The existing floor beam is made from an aluminum extrusion 184.1 mm (7.25 in.) deep. The top and bottom chords are 50.8 mm (2.0 in.) by 3.17 mm (0.125 in.), and the web is 177.8 mm (7.0 in.) by 2.54 mm (0.100 in.)

The concept 1 floor beam design is shown in figure 4. The top and bottom chords of the existing floor beam extrusion are machined down in thickness and graphite is bonded in place. A thin metal strip is bonded to the surface of the graphite to produce a balanced design and minimize the thermal distortion. It also acts as an abrasion protection for the composite. The web is not modified in this design.

The concept 2-1 floor beam design is shown in figure 5. The chords are reinforced with graphite and the web consists of an aluminum-faced honeycomb sandwich. Where the cutouts exist in the web of the beam, metal doublers are bonded on as local reinforcement. Figure 6 shows the concept 2-2 floor beam design. This design is identical to concept 2-1 with the aluminum replaced with titanium.

The concept 3-1 floor beam design, shown in figure 7, is an all-graphite composite design with metal abrasion strips and angles bonded to the chords. The fabrication process considered for this design is to lay up the composite on a rectangular mandrel and then split the composite after curing. The two channel pieces are then bonded back-to-back to form the web of the beam. The web would have metal doublers bonded on for reinforcement around the cutouts. Figure 8 shows the concept 3-2 design, in which PRD-49 instead of graphite is used for the web material.

In each floor beam design, the seat tracks are attached by drilling through the composite in the chords. This composite hole-out material has not been considered as load-carrying material. The material could be replaced by a fiberglass strip, but to simplify fabrication the graphite is made continuous across the width of the chords. The ends of the floor beams are reinforced with metal doublers for attachment to the body frames.

The weight estimate and the weight of composite for each design are shown in table 5.

TABLE 5.—FLOOR BEAM WEIGHT SUMMARY

Concept	Description	Weight		Weight saving			Composite weight	
		kg	lbm	kg	lbm	%	kg	lbm
Baseline	Existing 727-200	7.17	15.8	–	–	–	–	–
1	Aluminum-graphite chords, unchanged web	5.76	12.7	1.41	3.1	19.6	0.74	1.63
2-1	Aluminum-graphite chords, aluminum honeycomb web	4.85	10.7	2.32	5.1	32.3	0.74	1.63
2-2	Titanium-graphite chords, titanium honeycomb core	4.76	10.5	2.41	5.3	33.6	0.74	1.63
3-1	Aluminum-graphite chords, ±45° graphite web	6.03	13.3	1.14	2.5	15.8	4.17	9.2
3-2	Aluminum-graphite chords, ±45° PRD-49 web	5.53	12.2	1.64	3.6	22.8	3.04 0.74	6.7[a] 1.63[b]

[a] PRD-49 [b] Graphite

3.4.2 Floor Panels

The existing 727-200 floor panels in the underseat area consist of aluminum faces on PVC core at a weight of 3.37 kg/sq m (0.69 lbm/sq ft). The floor panels in the aisle area consist of aluminum faces on balsa wood core at a weight of 4.05 kg/sq m (0.83 lbm/sq ft). Rolls-Royce Composite Materials Limited has designed all-graphite composite floor panels which are suitable for replacement of the existing panels. These panels have been included in the applications study to demonstrate the typical weight savings available by using all-composite floor panels. The design of the composite floor panels is shown in figure 9.

The composite floor panels that would replace those in the underseat area weigh 2.34 kg/sq m (0.48 lbm/sq ft) and the composite aisle floor panels weigh 2.64 kg/sq m (0.54 lbm/sq ft). The weight saved for a 4.57-m (180-in.) length of fuselage is 21.3 kg (47 lbm), which is a 40% savings. The total floor panel composite material weight for the fuselage test section is 11.6 kg (25.6 lbm).

3.4.3 Keel Beam

The keel beam in the 727-200 forms the continuous compression member in the lower quadrant between body stations (BS) 740 and 950. The keel beam load in the fuselage section forward of BS 740 is distributed to adjacent stringers between BS 640 and 740. The keel beam in the forebody section, as shown in figure 10, consists of two identical aluminum machined extrusions located at stringer (STR) 28 on the left and right side of the aircraft bottom centerline. Each keel beam section contains approximately 968 sq mm (1.5 sq in.) of effective material at BS 740.

The design of the reinforced aluminum keel beam is shown in figure 11. This design consists of two identical machined aluminum sections bonded with layers of graphite/epoxy. The layers of graphite terminate on a stack of vertical titanium step fittings at BS 740. The number of layers of graphite decreases from BS 740 to 680 to conform to the load reduction caused by shear redistribution. A cap strip is bonded across the top of the section to prevent delamination of the graphite layers caused by flexing of the skin. The reinforced keel beam assembly is mechanically fastened to the panel skin.

The existing keel beam elements weigh 22 kg (48 lbm). The graphite-composite-reinforced aluminum keel beam weighs 14 kg (30 lbm), with a weight savings of 37.5% and a composite weight of 2.7 kg (6.0 lbm).

A titanium keel beam design was developed for the concept 2 lower quadrant panel. The design is identical to the reinforced aluminum beam with titanium replacing the aluminum sections. The weight of this reinforced titanium keel beam is 14 kg (31 lbm) for a weight savings of 35.4% and a composite weight of 2.7 kg 6.0 lbm).

3.4.4 Upper Quadrant—Tension Skin Panel

The upper quadrant skin panel, shown in figure 12, extends from BS 680 to 740 and from STR 10 left to STR 10 right. The panel consists of 2024-T3 machine-tapered skins and 7075-T6 tapered stringers. The stringers are riveted to the skins, and the panel assembly is attached to the body frames by shear clips at each frame and stringer intersection. Fail-safe straps are bonded to the skin at each frame location. The skin gage varies from a minimum of 1.01 mm (0.040 in.) to a maximum of 1.05 mm (0.065 in.). The skin gage between STR 10 and 4 at BS 740 increases to 2.54 mm (0.100 in.) thickness to account for high panel shears caused by body bending and shear redistribution from the BS 740 bulkhead.

The reinforced aluminum concept 1 design, shown in figures 13 and 14, consists of a uniform 0.91-mm (0.036-in.) thick 2024-T3 aluminum skin with a 0.63-mm (0.025-in.) thick 2024-T3 bonded waffle doubler and graphite-reinforced, riveted aluminum stringers. The skin panel assembly is attached to the body frames in the same manner as the existing structure. The graphite reinforcing is bonded on both sides of each outstanding leg of the stringer as shown in figure 14. Each composite layer at BS 680 and 740 is bonded to a titanium step fitting. The stringers are 0.63-mm (0.025-in.) constant thickness formed from 7075-T6 aluminum sheet material. The graphite tapers in thickness from a maximum at BS 740 to a minimum at BS 680 to accommodate the reduction in the bending moment.

During bonding the residual thermal stresses are reduced by restraining the aluminum stringer on a steel tool base. This method of fabrication will produce a stress-free temperature at approximately 322° K (120° F) for a 394° K (250° F) cure cycle. The resulting residual thermal stress at 218° K (-67° F) plus the operating fatigue stresses are not severe enough to cause premature fatigue damage.

The weight of the existing upper quadrant panel including the skin, stringers, body frames, and shear clips is 144 kg (317 lbm). The concept 1 weight for the same structural elements is 115 kg (254 lbm). This results in a weight savings of 19.8% with a composite weight of 6.8 kg (15.0 lbm).

The reinforced titanium concept 2 design, shown in figures 15 and 16, consists of a uniform 0.381-mm (0.015-in.) 6Al-4V titanium skin reinforced with one ply of graphite at 90° for pressure load and two plies of graphite at 45° for shear. An additional doubler of 0.254-mm (0.010-in.) titanium is bonded to the skin along BS 740, and a ±45° graphite doubler is added at BS 740 between STR 7 and 10. The stringers are of constant thickness formed from 0.406-mm (0.016-in.) 6Al-4V titanium sheet. The graphite reinforcing is bonded to the top of the hat section, and the stringer is bonded to the skin panel on top of the graphite reinforcing. The graphite layers on the stringers taper in thickness to accommodate the varying bending moment. Titanium frame tees are bonded to the skin panel with cutouts for each stringer. The bonded portion of the frame tee is continuous and acts as a fail-safe strap. J-section titanium frames are mechanically attached to the frame tee segments, and flanged cutouts in the frame webs accommodate the stringers. Lateral stability of the body frames in the upper quadrant is provided with five tension ties at approximately 760-mm (30-in.) spacing.

The panel contains mechanical splices at STR 1, 4, and 7. These splices are required because of the limited width of titanium sheet. Each 760-mm (30-in.) wide panel section contains a picture frame titanium step fitting for the composite reinforcing. The panel sections are riveted together with titanium flush head rivets. A 0.254-mm (0.010-in.) thick strip of titanium is bonded along each mechanical splice to prevent knife edging of the countersunk fasteners. The bonded frame tee portions are also spliced at each longitudinal splice, and the frame web and inner chord are continuous from STR 10 right to STR 10 left.

The weight of the concept 2 design including the skin panel, stringers, frame tees, and frame J-section is 113 kg (248 lbm). This results in a weight savings of 21.7% with a composite weight of 14.4 kg (31.7 lbm).

The all-composite concept 3 design, shown in figures 17 and 18, consists of cross-ply-laminated graphite skins and fiberglass graphite stringers. The axial load ratio between the stringers and skin is similar to that of the existing structure. The exterior of the panel contains a bonded, fine wire mesh screen for lightning protection and static dissipation. The basic skin contains three plies of graphite at 90° for pressure load and two plies at 45° for shear. The axial load in the skin is taken by 0° plies tailored to the varying load levels. The skin contains additional plies at ±45° in high-shear areas.

The stringers are molded fiberglass hat sections with uniaxial graphite reinforcing. The fiberglass is considered to be a shear tie for the graphite reinforcing. The thickness of the fiberglass section was based on compression crippling calculations. Fiberglass was chosen for this application because of its thermal expansion compatibility with graphite composite and the availability of standard fabrication processes. All-graphite composite stringers were not considered because of the labor costs associated with present hand layup fabrication techniques. If advanced methods of manufacturing of structural sections in graphite composite are available to support the developmental program, then the fiberglass stringers would be replaced with graphite composite sections.

The construction of body frames and method of supplying lateral stability are identical to the concept 2 design. There are no separate fail-safe straps in the panel, since results from Boeing IR&D tests indicate that the three layers of 90° plies of graphite together with the cross-plied ±45° layers are able to contain local damage. The layers of graphite in the skin are continuous across the entire surface of the panel. Each layer terminates on a titanium step picture frame which surrounds the panel.

The area between STR 9 and 10 will contain a transition from skin stringer to honeycomb to match the side quadrant panel. The panel splice at STR 10 will be a typical honeycomb double-lap splice joint. These details are shown in figure 18.

The weight of the concept 3 design for the skin panel, stringers, frame tees, and frame J-section is 111 kg (245 lbm). This results in a weight savings of 22.7% with a composite weight of 47.6 kg (105 lbm).

The weights of all three upper quadrant designs are summarized in table 6.

TABLE 6.—UPPER QUADRANT PANEL WEIGHT SUMMARY

Concept	Description	Weight		Weight saving			Composite weight	
		kg	lbm	kg	lbm	%	kg	lbm
Baseline	Existing 727-200	144	317	—	—	—	—	—
1	Aluminum skin; aluminum-graphite stringer	115	254	29	63	19.8	6.8	15.0
2	Titanium reinforced skin; titanium-graphite stringer	112	248	32	69	21.7	14.4	31.7
3	Graphite skins; fiberglass-graphite stringers	111	245	33	72	22.7	47.6	105.0

3.4.5 Lower Quadrant—Compression Skin Panel

The lower quadrant skin panel, shown in figure 19, extends from BS 680 to 740 and from STR 19 left to STR 19 right. The panel contains 2024-T3 machine-tapered skins and 7075-T6 tapered stringers. The stringers are riveted to the skin, and the panel assembly is attached to the body frames by full-depth shear clip angles along each frame. Fail-safe straps are bonded to the skin at each frame location. The skin gage varies from a minimum of 1.02 mm (0.040 in.) to a maximum of 4.06 mm (0.160 in.) at BS 740 under the keel beam.

The reinforced aluminum concept 1 design, shown in figures 20 and 21, consists of a 2024-T3 aluminum machine-tapered skin with a bonded waffle doubler and graphite-reinforced, riveted aluminum stringers. The machined skin varies in thickness from 0.91 mm (0.036 in.) to 3.81 mm (0.150 in.). The 2024-T3 aluminum bonded waffle doubler covers the skin area where the thickness is less than 1.14 mm (0.045 in.). The skin panel assembly is attached to the body frames with full-depth shear clip angles along each frame. The stringers are inverted hat sections with graphite composite bonded to the top surface. The number of layers of graphite is tailored to match the variation in the compression end load. Each layer of graphite at BS 680 and 740 terminates on a titanium step fitting. The stringers are of constant thickness fabricated from 0.91-mm (0.036-in.) and 1.02-mm (0.040-in.) gage material. The gage of both stringer types was based on compression crippling calculations. The longitudinal portion of the waffle doubler between the fail-safe strap locations and between the rivet lines is chemically milled to reduce weight. This detail is shown in figure 21. The stringers have been inverted to increase the effective width of the skin under compression load, which results in a reduction of the overall skin thickness. The composite-reinforced aluminum keel beam chords previously defined are mechanically attached along the STR 28 location. The longitudinal mechanical splices at STR 19 and 26 are the same as those used on the existing structure.

Residual thermal stress alleviation processes will be used for fabrication of the stringers in this panel to reduce distortion of the parts and simplify fastener installations. The test results presented in reference 2 indicate that residual thermal tension stresses improve the buckling capacity of compression members. This additional compression capacity has not been accounted for in these preliminary designs but will be accounted for when the designs are refined.

The weight of the existing lower quadrant panel including the skin, stringers, body frames, shear clip angles, and the keel beam chords is 190 kg (419 lbm). The concept 1 weight for the same structural elements is 157 kg (346 lbm). This results in a weight savings of 17.4% with a total composite weight of 7.3 kg (16.2 lbm).

The reinforced titanium concept 2 design for the lower quadrant, shown in figures 22 and 23, consists of 6Al-4V titanium skins reinforced with one ply of graphite at 90° for pressure and two plies of graphite at 45° for shear and graphite-reinforced titanium stringers. The skin panel between STR 19 and 23 is 0.51-mm (0.020-in.) constant thickness and the skin panel between STR 23 and 27 is 0.46-mm (0.018-in.) constant thickness. An additional doubler is bonded to this panel section between STR 26 and 27 at BS 740 for additional shear capacity. The skin panel between STR 27 left and right is 0.71-mm (0.028-in.) constant thickness with a tapered doubler bonded to the panel. There is no graphite reinforcing on

this skin panel because of the mechanical attachment requirements of the keel beam chords. The stringers are of constant thickness formed from 6Al-4V titanium sheet that varies in gage from 0.64 mm (0.025 in.) to 1.02 mm (0.040 in.). The gage of each stringer was based on compression crippling calculations. The graphite reinforcing is bonded to the top of the hat section and the stringer is bonded to the skin panel on top of the graphite reinforcing. The graphite layers on the stringers taper in thickness to accommodate the variations in the compression end load. Bonded titanium frame tees, titanium J-section frames, and frame lateral stability straps are assembled on the skin panel in a similar manner to that described for the top quadrant concept 2 panel. The mechanical longitudinal skin splices are also identical to those described for the top quadrant design.

The keel beam chords are mechanically fastened to the skin panel. Bonding was not considered because of the possibility of creating bond voids due to the relative stiffness of the chord section.

The weight of the concept 2 design for the skin panel, stringers, frame tees, frame sections, and the keel beam chords is 155 kg (341 lbm). This results in a weight savings of 18.6% with a composite weight of 18.7 kg (41.3 lbm).

The all-composite concept 3 design, shown in figures 24 and 25, consists of cross-ply-laminated graphite skins on 19.05-mm (0.75-in.) thick aluminum honeycomb core. The exterior skin contains a bonded, fine wire mesh screen for lightning protection and static dissipation. The inner skin contains two plies at 90° for pressure, two plies at 45° for shear, and one ply at 0° for end load. The outer skin contains one ply at 90°, two plies at 45°, and one ply at 0°. Plies at 0° are bonded to the panel in the keel beam area for additional end load, and ±45° graphite doublers are added for increased shear capacity in the keel beam shear redistribution area. Frame tees are bonded fiberglass sections, and titanium frame J-sections are mechanically attached. As in the upper quadrant design, there are no separate fail-safe straps. The basic plies of graphite in the skin are continuous across the entire surface of the panel. Each basic ply terminates on a titanium step picture frame which surrounds the panel. The panel splice at STR 19 location is a honeycomb-type double-lap splice.

The keel beam load at STR 28 is introduced into a U-shaped titanium fitting, which extends 1520 mm (60 in.) forward of BS 740 and contains machined load transfer steps that shear the compression load through the ±45° doublers and into the 0° plies. The skin between STR 28 left and right is a single titanium sheet mechanically fastened to the edges of the titanium fitting.

The weight of the concept 3 design for the skin panel, frame tees, frame J-section, and the titanium keel beam fitting is 152 kg (335 lbm). This results in a weight savings of 20.0% with a composite weight of 47.4 kg (104.6 lbm).

The weight of all three design concepts for the lower panel is summarized in table 7.

TABLE 7.—LOWER QUADRANT PANEL WEIGHT SUMMARY

Concept	Description	Weight		Weight saving			Composite weight	
		kg	lbm	kg	lbm	%	kg	lbm
Baseline	Existing 727-200	190	419	—	—	—	—	—
1	Aluminum skin; aluminum-graphite stringer	157	346	33	73	17.4	7.3	16.2
2	Titanium reinforced skin; titanium-graphite stringer	155	341	35	78	18.6	18.7	41.3
3	Graphite skins; aluminum honeycomb core	152	335	38	84	20.0	47.4	104.6

3.4.6 Side Quadrant—Shear Skin Panel

The side quadrant skin panel, shown in figure 26, extends from BS 680 to 740 and from STR 10 to 19. The panel contains 2024-T3 machine-tapered skins, 7075-T6 tapered stringers, and 7075-T73 forged window frames. The stringers and window frames are riveted to the skin, and the panel assembly is attached to the body frames by full-depth shear clip angles along each frame. Fail-safe straps are bonded to the skin at each frame location. The skin gage varies from a minimum of 1.02 mm (0.040 in.) to a maximum of 6.10 mm (0.240 in.) at BS 740 in the window area. The panel has a wing scanning light cutout at STR 15 between BS 720B and 720C.

The reinforced aluminum concept 1 design, shown in figures 27 and 28, consists of 2024-T3 aluminum machine-tapered skins, 2024-T3 bonded waffle doublers, 7075-T73 forged window frames, and 7075-T6 graphite-reinforced stringers. The skin panel between STR 10 and 14 varies from 1.52 mm (0.060 in.) to 4.06 mm (0.160 in.) thick and the bonded waffle doubler in this area is 0.81 mm (0.032 in.) thick. The skin panel between STR 14 and 19 varies in thickness from 1.02 mm (0.040 in.) to 2.41 mm (0.095 in.) and the doubler in this area is 0.63 mm (0.025 in.) thick. The skin panel assembly is attached to the body frames with full-depth shear clip angles along each frame.

The splice stringers at STR 10, 14, and 19 are similar to those in the upper quadrant, with graphite reinforcing bonded to the outstanding legs. Stringers STR 11 and 13 are Z-sections with graphite reinforcing, and stringers STR 15, 16, and 18 are similar to those in the lower quadrant. The crease beam stringer at STR 17 was not designed with graphite reinforcing because of mechanical attachment requirements of the floor structure. The doubler between STR 10 and 14 is profiled around the window frame to carry the pressure load and shear around the cutout.

Residual thermal stress alleviation processes would be used for fabrication of all the stringers to reduce distortion and improve the fatigue properties.

The weight of both existing side quadrant panels including the skin panel, stringers, body frames, and shear clips is 211 kg (465 lbm). The concept 1 weight for the same structural elements is 171 kg (377 lbm). This results in a weight savings of 18.9% with a composite weight of 4.1 kg (9.0 lbm).

The reinforced titanium concept 2 design for the side quadrant, shown in figures 29 and 30, consists of graphite-reinforced 6Al-4V titanium skins and graphite-reinforced titanium stringers. The skin panel between STR 10 and 14 is taper machined from a nominal thickness of 0.76 mm (0.030 in.) to 3.17 mm (0.125 in.). The panel contains machined pockets for composite reinforcing. The area between STR 10 and 11 contains two plies at 90° in the recesses and the area between STR 11 and 13 contains two plies at 45°. The area between STR 13 and 14 contains two plies at 45° in the recesses. Two plies at 45° are also laid over the entire area from STR 10 to 14. The plies in the recesses are bonded to steps machined in the basic skin, and the overall ±45° plies are bonded to step fittings located around each window and along the splices at STR 10 and 14. The skin panels between STR 14 and 19 are of uniform thickness 0.38 mm (0.015 in.), with one ply at 90° and two plies at 45°. A titanium doubler is added at BS 740 and STR 17, and ±45° graphite doublers are added in other areas of high shear. The stringers are of constant thickness formed from 6Al-4V titanium sheet or machined from extrusions. The stringer sections used in this design are shown in figure 30. The crease beam, STR 17, was not designed for composite application because of the panel splice and floor structure mechanical attachment requirements. The window frame is machined from titanium and bonded to the skin panel. Bonded titanium frame tees, titanium J-section frames, and frame lateral stability straps are assembled on the skin panel in a manner similar to that described for the upper and lower quadrant concept 2 panels. The mechanical longitudinal skin splices are also identical to those described for the upper and lower quadrant designs.

The weight of the concept 2 design for both side panels is 144 kg (318 lbm). This results in a weight savings of 31.6% with a composite weight of 10.0 kg (22.0 lbm).

The all-composite concept 3 design, shown in figures 31 and 32, consists of cross-ply-laminated graphite skins on 19.1-mm (0.75-in.) thick aluminum honeycomb core. The exterior skin contains a bonded lightning protection screen similar to that of the upper and lower quadrant panels. The basic inner skin contains two plies at 90°, two plies at 45°, and one ply at 0°. The basic outer skin contains one ply at 90°, two plies at 45°, and one ply at 0°. Additional plies at 90° are bonded to both skins between STR 10 and 14. Also, additional ±45° plies are bonded in this area. The large end load at the STR 17 location is taken by a series of 0° plies on both inner and outer skins. The floor structure is connected to the body shell by means of a fiberglass tee bonded to the panel surface. The window frames are machined titanium sections that contain bond load transfer steps for each ply. Frame tees are bonded fiberglass sections, and titanium frame J-sections are mechanically attached. As in the upper and lower quadrant design, there are no separate fail-safe straps. The basic plies of graphite in the skin are continuous across the entire surface of the panel. Each basic ply terminates on a titanium step picture frame which surrounds the panel.

The weight of the concept 3 design for both side panels is 137 kg (302 lbm). This results in a weight savings of 35.0% with a composite weight of 39 kg (86 lbm).

The weights of all three design concepts for the side panels are summarized in table 8.

TABLE 8.—SIDE QUADRANT PANEL WEIGHT SUMMARY

Concept	Description	Weight		Weight saving			Composite weight	
		kg	lbm	kg	lbm	%	kg	lbm
Baseline	Existing 727-200	211	465	–	–	–	–	–
1	Aluminum skin; aluminum-graphite stringer	171	377	40	88	18.9	4.1	9
2	Titanium reinforced skin; titanium-graphite stringer	144	318	67	147	31.6	10.0	22
3	Graphite skins; aluminum honeycomb core	137	302	74	163	35.0	39.0	86

The weight of a fuselage section containing each of the three design concepts is shown in table 9. The all-composite floor panels are included in the weights for each of the three concepts; the concept 1 weight incorporates concept 2-1 floor beams, and the concept 2 and 3 weights, concept 2-2 floor beams.

TABLE 9.—WEIGHT SUMMARY OF FUSELAGE SECTION

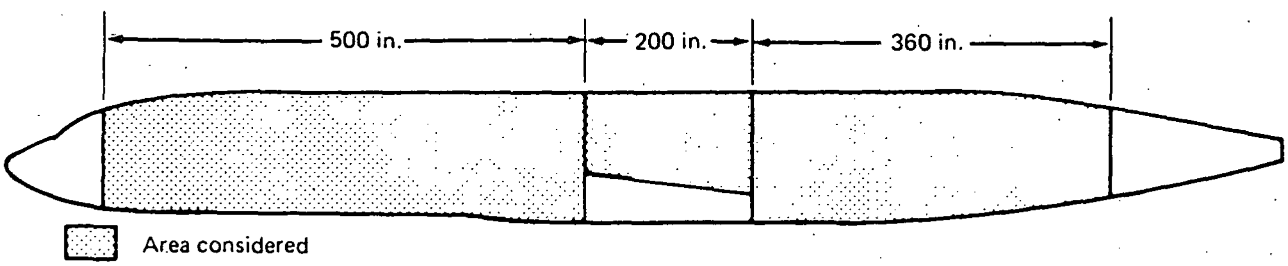
Component	Baseline		Concept 1		Concept 2		Concept 3	
	kg	lbm	kg	lbm	kg	lbm	kg	lbm
Shell								
Upper quadrant	144	317	115	254	112	248	111	245
Side quadrant	211	465	171	377	144	318	137	302
Lower quadrant	168	371	143	316	141	310	152	335
Floor beams	57	126	39	86	38	84	38	84
Floor panels	53	116	31	69	31	69	31	69
Keel beam	22	48	14	30	14	31	(a)	(a)
Total weight	655	1443	513	1132	480	1060	469	1035
Weight saving	–	–	141	311	175	383	186	408
% weight saving	–	–	21.5	21.5	26.5	26.5	28.2	28.2
Weight of composite	–	–	34.4	75.8	60.6	133.6	166.3	366.6
CEF[b]	–		4.10		2.87		1.11	

[a]Included in lower quadrant panel
[b]Composite efficiency factor

3.5 COST BENEFIT STUDIES

Weight savings for an entire fuselage section between the flight deck area and the aft pressure bulkhead were obtained, for each of the concepts studied, by extrapolating the savings in the study section to the total fuselage on the basis of primary structural weight and bending moments and shear loads. The results are shown in table 10. This table also contains weight estimates of the systems impact of the concept 2 and 3 designs which will result from significant changes to electromagnetic interference and electrical ground paths. The large reduction in the weight savings for the total fuselage as compared to the study section is due to the fact that the large bulkheads at the front and rear spars are included in the total primary structural weight and no weight savings were obtained for these components in the applications study. The weight savings for the total fuselage were further reduced because of the large area of minimum-gage structure outside of the study section.

TABLE 10.—TOTAL FUSELAGE WEIGHT SAVING SUMMARY



Concept	Weight		Structural impact		System impact			CEF
	kg	lbm	kg	lbm	kg	lbm	%	
Baseline	5195	11 452	–	–	–	–	–	–
1	4794	10 569	-401	- 883	–	–	7.8	1.25
2	4511	9 944	-684	-1508	+ 45.4	+100	12.2	1.12
3	4465	9 843	-730	-1609	+136	+300	11.4	0.34

Manufacturing production methods for each design concept were formulated, and cost estimates were developed for a 300-airplane production program at a cost for graphite composite of $132/kg ($60/lbm). These cost estimates, shown in table 11, do not include major costs for facilities. The effect on the cost per pound saving of reducing the composite price is shown in figure 33.

TABLE 11.–WEIGHT SAVING AND INCREASED PRODUCTION COSTS FOR TOTAL FUSELAGE

Item	Units	Baseline	Concept 1	Concept 2	Concept 3
Total primary fuselage weight	kg	5 195	4 794	4 511	4 465
	lbm	11 452	10 569	9 944	9 843
Weight saved[a]	kg	–	401	684	730
	lbm	–	833	1 508	1 609
Increased production cost[b]	$	–	27 786	131 130	160 875
Cost per kilogram (pound) of saving	$/kg	–	69.30	191.70	220.30
	$/lbm	–	31.50	87.00	100.00
Composite weight in fuselage	kg	–	321	568	1 725
	lbm	–	707	1 253	3 804

[a]Structural weight saving

[b]Graphite composite cost at $132/kg ($60/lb)

An economic analysis was performed to define the cost benefits to airline companies of incorporating composite structures on commercial jet aircraft. The analysis was applied to the total 727-200 fuselage using the weights and costs shown in table 11 and the following set of conditions.

- The average trip length and yearly utilization rate was obtained from 1969 U.S. domestic trunk route data.
- The aircraft was considered to have the existing payload capability and number and type of engines.
- The total aircraft weight savings was taken as twice that saved for the fuselage alone. This cascading ratio has been substantiated by Boeing IR&D studies.
- Fuel and maintenance costs were the only two items credited as direct operating cost savings. The fuel savings were obtained from aircraft performance curves, and the reduction in maintenance costs was calculated from the following Air Transport Association equation with 1971 coefficients.

$$\text{Maintenance hours per flight hour} = \frac{0.01\ Wa}{1000} + 6 - \frac{680}{(Wa/1000) + 85}$$

where Wa = airframe weight.

- Direct labor costs plus overhead were computed at $15/hr.
- Aircraft insurance costs were increased due to the more expensive construction.
- The present values of cost savings and insurance were based on a 15% cost of capital for 12 years of service.

The results of this analysis, shown in table 12, indicate that only concept 1 is cost effective at a graphite composite price of $132/kg ($60/lbm). The analysis was further extended to show the effect of reducing the composite cost to $44/kg ($20/lbm). The savings in fuel and maintenance were kept constant, and the insurance costs were reduced according to the reduction in the production costs. The results, shown in figure 34, indicate that concept 1 rapidly increases in cost effectiveness, and concept 3 becomes cost effective at a composite price of approximately $77/kg ($35/lbm). The projected cost estimates shown in figure 35 indicate that concept 3 will become cost effective in late 1976. The results shown in figure 34 indicate that concept 2 would not become cost effective until the graphite composite price is less than $44/kg ($20/lbm). The effect of a reduction in the material cost on this concept was not sufficient to offset the estimated increased production costs.

TABLE 12.—ECONOMIC ANALYSIS FOR CASCADED WEIGHT SAVINGS OF A 727-200 TOTAL AIRCRAFT

Item	Units	Concept 1	Concept 2	Concept 3
Weight saved, total aircraft	kg	801	1 368	1 460
	lbm	1 766	3 016	3 218
Increased production cost	$	27 786	131 130	160 875
Cost per kilogram (pound) saved	$/kg	34.70	95.80	110.0
	$/lbm	15.73	43.47	50.00
Present value of fuel savings	$	19 980	34 139	36 404
Present value maintenance reduction	$	14 750	25 636	27 280
Present value of insurance	$	−1507	−7097	−8726
Summation of present values	$	33 223	52 678	54 958
Present value/increased cost	–	1.19	0.40	0.34

A detailed economic analysis to include the benefits of increased revenue was not performed, since this type of analysis requires a definition of route structures and passenger load factor. A brief study of airline routes and recorded data of passenger load factor indicates that the use of composites on commercial airlines will have a present value of between $22/kg ($10/lbm) and $331/kg ($150/lbm) of weight saving. The higher present values apply to a very limited number of routes involving the use of very few aircraft.

3.6 DEVELOPMENTAL PROGRAM COMPONENTS AND RELATIVE COSTS

Developmental program costs were estimated, and the relative cost for each concept is defined as follows:

Concept	Relative Developmental Program Cost
1	1.0
2	2.4
3	3.3

The concept 1 program cost includes the developmental program plan as defined in section 4.0 and appendix A and two flight service evaluation aircraft. Each service evaluation aircraft will contain a top quadrant panel and a lower quadrant panel. The top quadrant panel will provide information on tension- and tension-fatigue-loaded structure, and the lower quadrant panel will provide information on compression-loaded structure. The top quadrant panel has been extended beyond the original study section, since this panel size is better suited for the aircraft assembly. Also, this extended panel will evaluate uniaxial composite reinforcing through a wider range of end loads. The composite-reinforced side quadrant panels will not be installed on the flight evaluation aircraft, since the axial stress levels in the reinforced stringers are less than the tension and compression stresses experienced by the upper and lower quadrant stringers. It was considered that the additional information obtained by service evaluation of the side quadrant panels did not warrant the relatively high cost of the redesigned panels.

The concept 1 flight evaluation aircraft is shown in figure 36. The section selected will contain the following components:

- Concept 1—upper quadrant panel, 9.65 m (380 in.) long
- Concept 1—lower quadrant panel, 4.57 m (180 in.) long
- Concept 2-1 floor beams, 8
- All-composite floor panels—4.57-m (180-in.) long section

The weights of the components in the concept 1 service evaluation aircraft are defined in table 13.

TABLE 13.–WEIGHT SUMMARY OF THE CONCEPT 1 FLIGHT EVALUATION COMPONENTS

Component	Baseline		Concept 1	
	kg	lbm	kg	lbm
Shell				
Upper quadrant, 9.65 m (380 in.) long	264	583	210	463
Lower quadrant, 4.57 m (180 in.) long	168	371	143	316
Floor beams	57	126	39	86
Floor panels	53	116	31	69
Keel beam	22	48	14	30
Total weight	564	1 244	437	964
Weight saving	–	–	127	280
% weight saving	–	–	22.5	22.5
Weight of composite	–	–	32.7	72.0

The concept 2 program cost includes the developmental program plan as defined in section 4.0 and appendix B and two flight service evaluation aircraft. The concept 3 program cost includes the developmental program plan as defined in section 4.0 and appendix C and two flight service evaluation aircraft. The flight evaluation aircraft for concepts 2 and 3 will contain: the respective 4.57-m (180-in.) long quadrant panels, eight concept 2-2 floor beams, and a 4.57-m (180-in.) long section of the all-composite floor panels. The weight of these components has been previously defined in table 9.

A cost comparison for installation of the flight components was made between inline production and retrofit installation. The inline production installation costs for all concepts were approximately one-half the retrofit costs; therefore, the inline production process was used for all concepts.

3.7 DISCUSSION OF STUDY RESULTS AND RECOMMENDATIONS

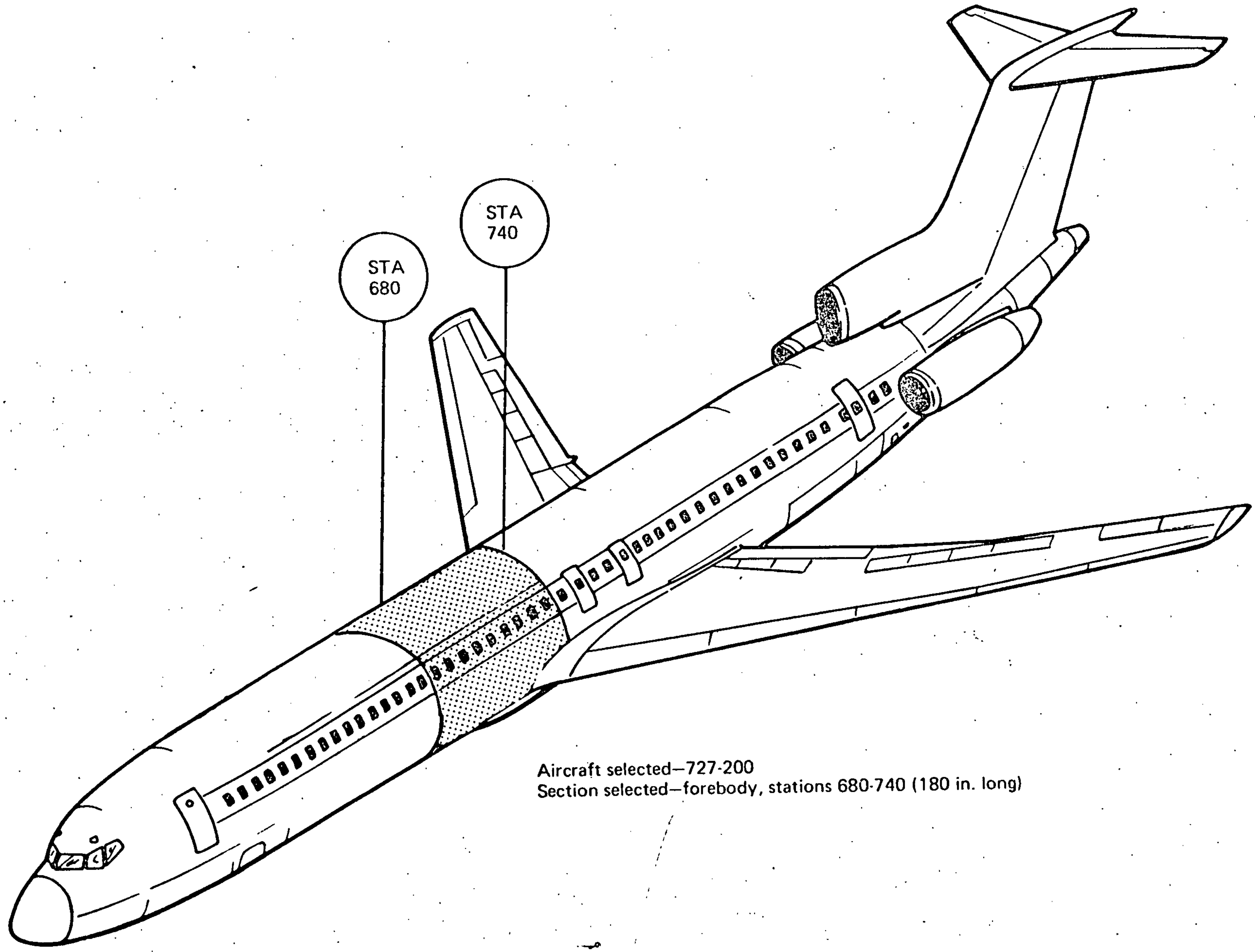
The weight saving estimates and cost benefits analysis produced during the application study were obtained for the aircraft section studied with the design constraints defined in section 3.4.

The concept 1 design represents a minimum developmental cost program. The results of the flight service evaluation program would provide valuable information on the effect of commercial aircraft service environment on uniaxial graphite-reinforced stringers in fuselage application. The results of the cost benefits study indicate that this design approach is presently cost effective, which will allow the concept to be incorporated in new aircraft designs and derivatives of present aircraft in the near future. A concept 1 airframe design will also result in minimum changes to inspection, maintenance, and electrical systems.

The results of the cost benefits study indicate that concept 2 is not cost effective based on the present fabrication cost estimates and projected graphite costs. However, the results from the concept 2 program would provide valuable information on the effect of commercial aircraft service environment on uniaxial reinforcement of metal structures and cross-ply composite laminates.

Implementation of the concept 3 developmental program would provide valuable information on the effect of commercial aircraft service environment on all-composite primary structure. The results of the cost benefits study indicate that the cost effectiveness of this concept is strongly influenced by the future graphite price. Incorporation of a concept 3 design into a commercial jet aircraft fuselage will require significant modifications to inspection, maintenance, and electrical systems. These changes will result in increased expenses for commercial airline companies, which were not included in the cost benefits study.

On the basis of the cost benefits study, the developmental program costs, and the potential for early implementation of composites on commercial jet aircraft, the concept 1 design is recommended for a 5-year flight service evaluation program.



Aircraft selected—727-200
Section selected—forebody, stations 680-740 (180 in. long)

FIGURE 3.—SELECTED AIRCRAFT AND FUSELAGE SECTION

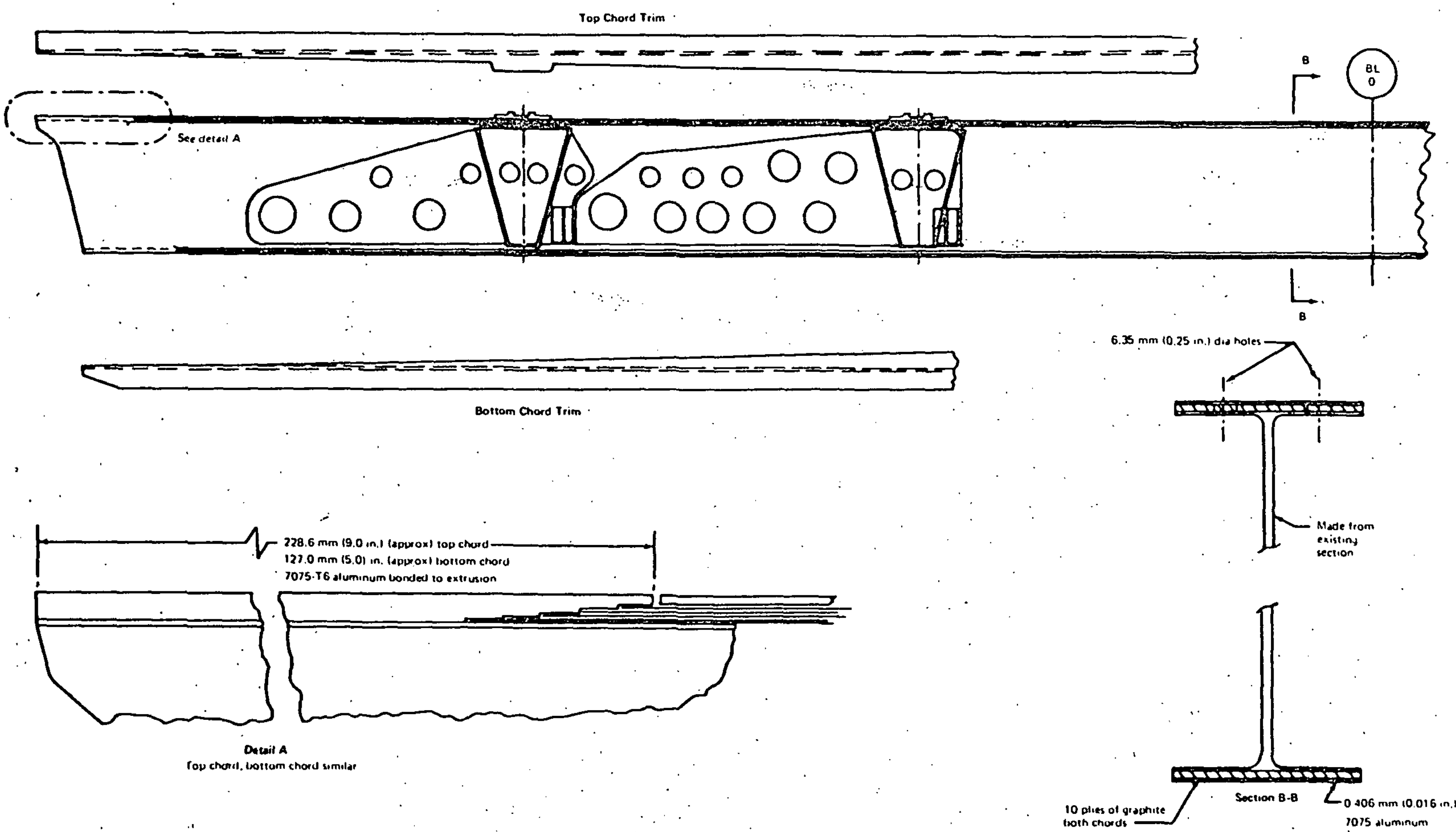


FIGURE 4.—CONCEPT 1 FLOOR BEAM AT STATION 720B

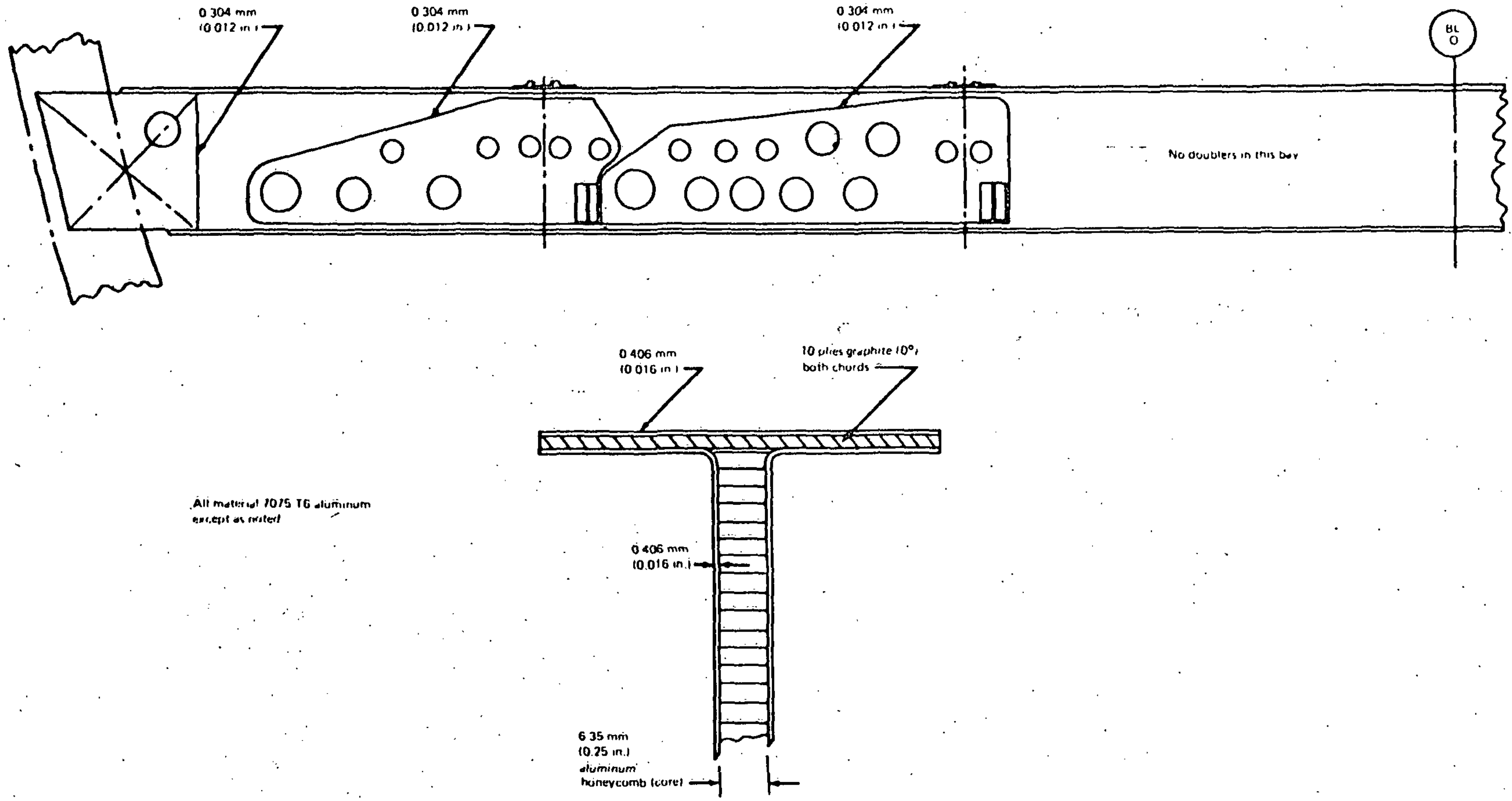


FIGURE 5.—CONCEPT 2-1 FLOOR BEAM AT STATION 720B

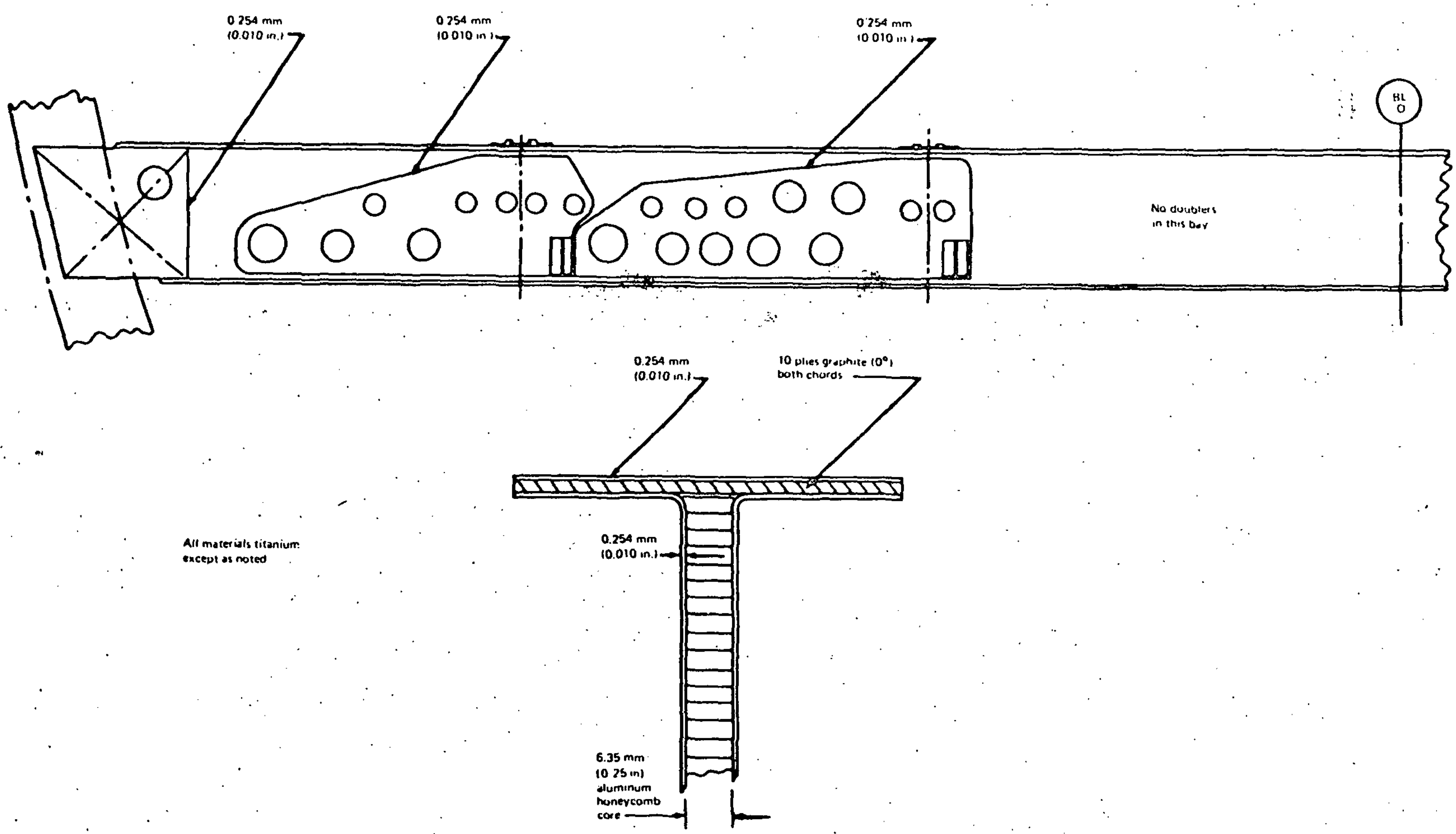


FIGURE 6.—CONCEPT 2-2 FLOOR BEAM AT STATION 720B

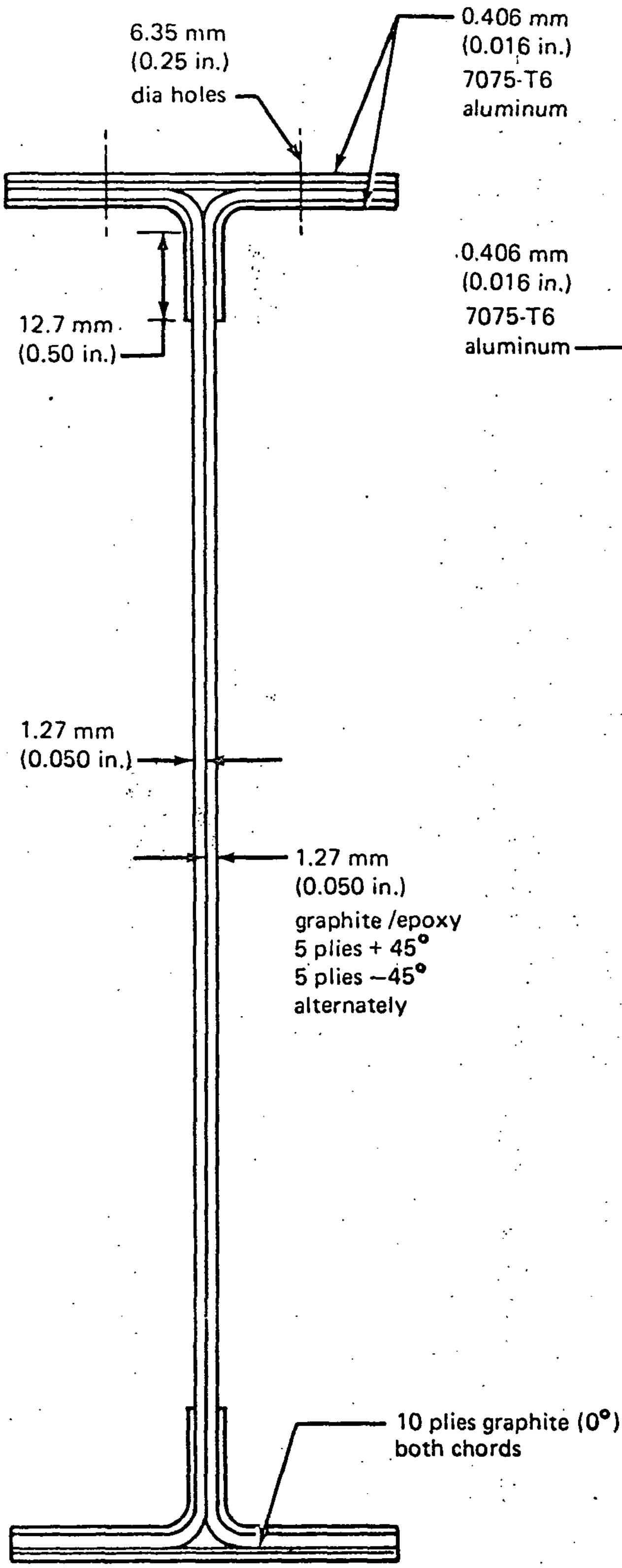


FIGURE 7.—CONCEPT 3-1 FLOOR BEAM AT STATION 720B

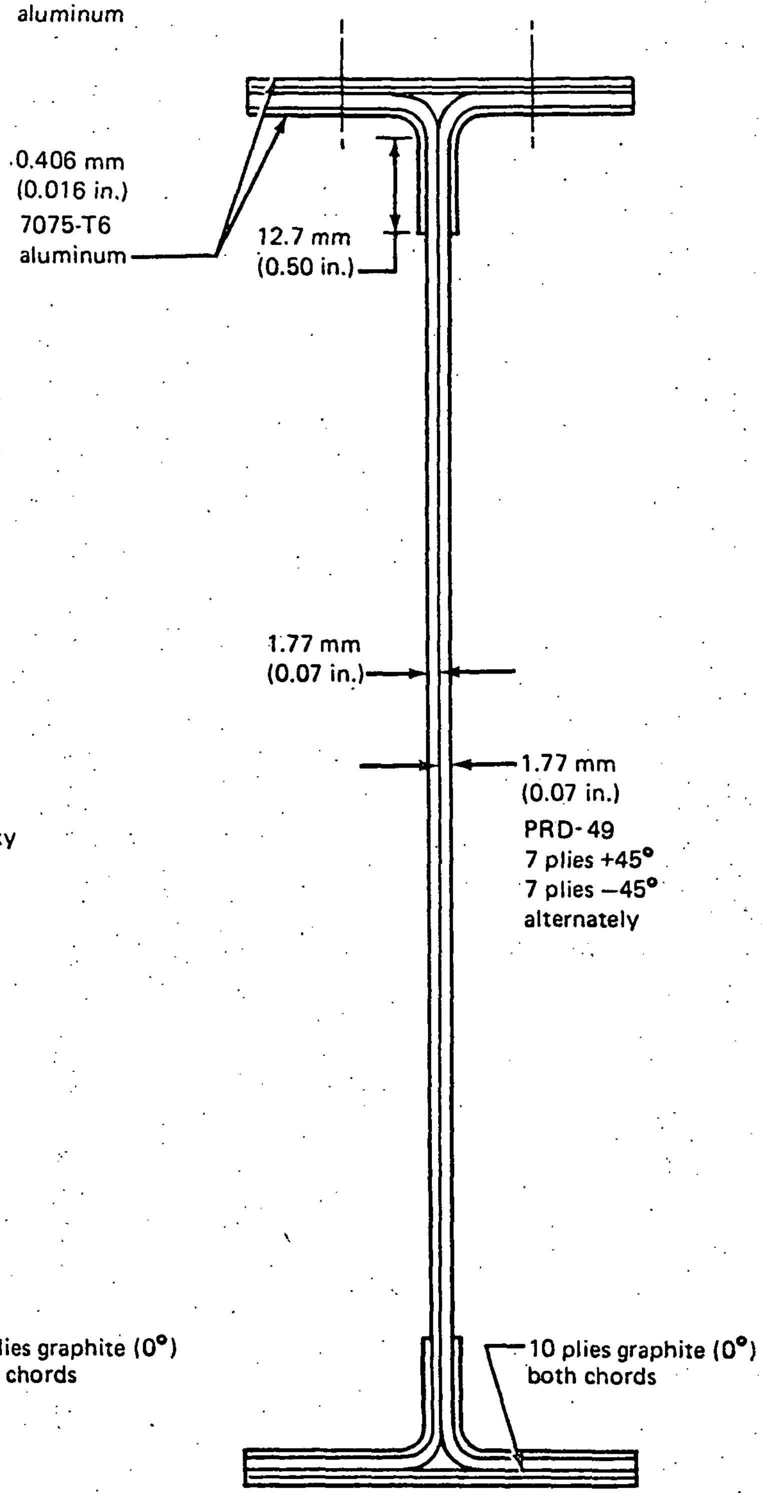


FIGURE 8.—CONCEPT 3-2 FLOOR BEAM AT STATION 720B

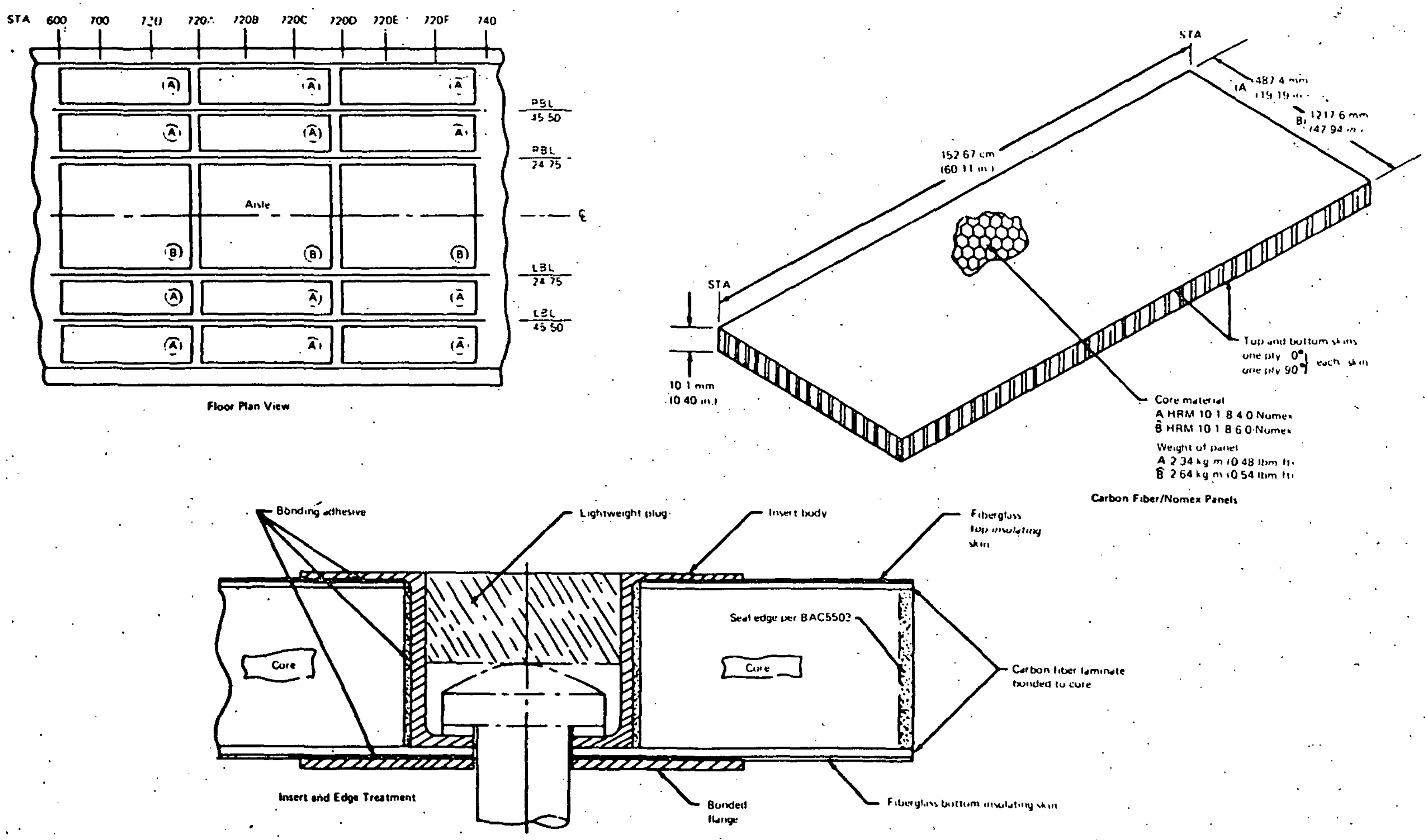


FIGURE 9.—FLOOR PANEL DESIGN

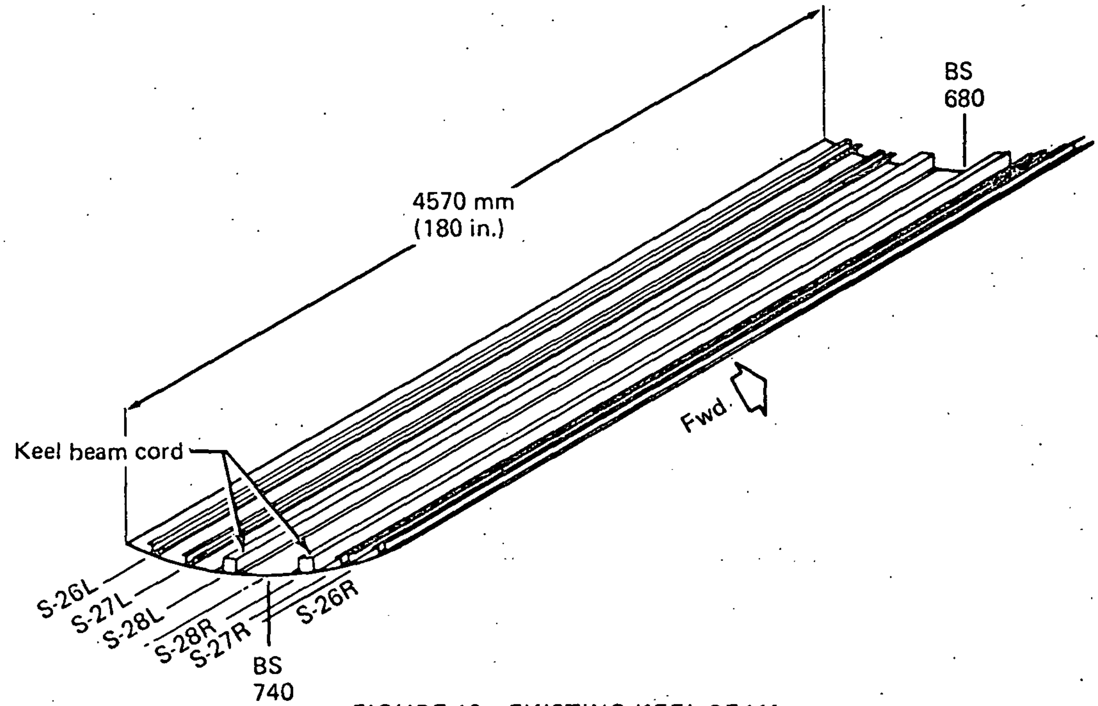


FIGURE 10.—EXISTING KEEL BEAM

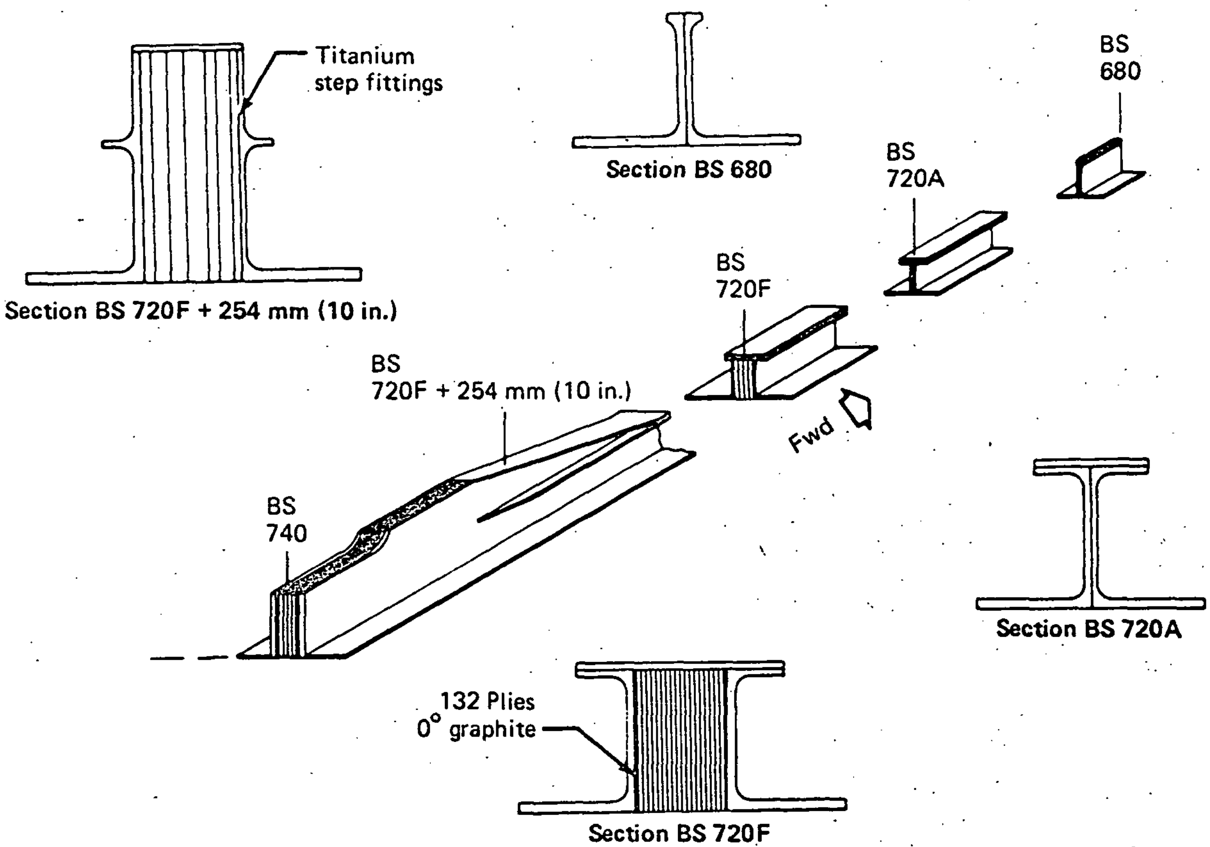


FIGURE 11.—ALUMINUM-REINFORCED KEEL BEAM DESIGN

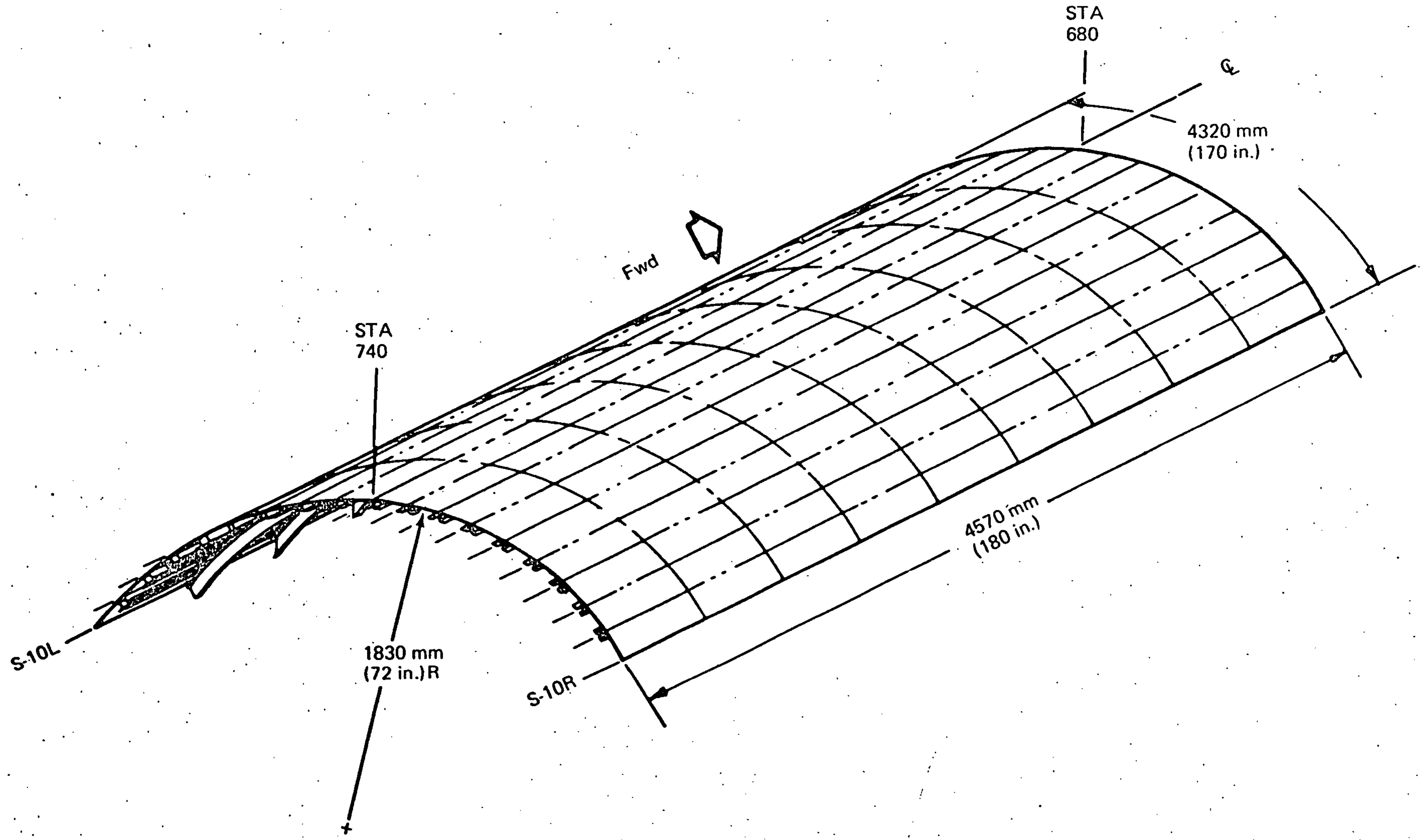


FIGURE 12.—EXISTING UPPER QUADRANT PANEL

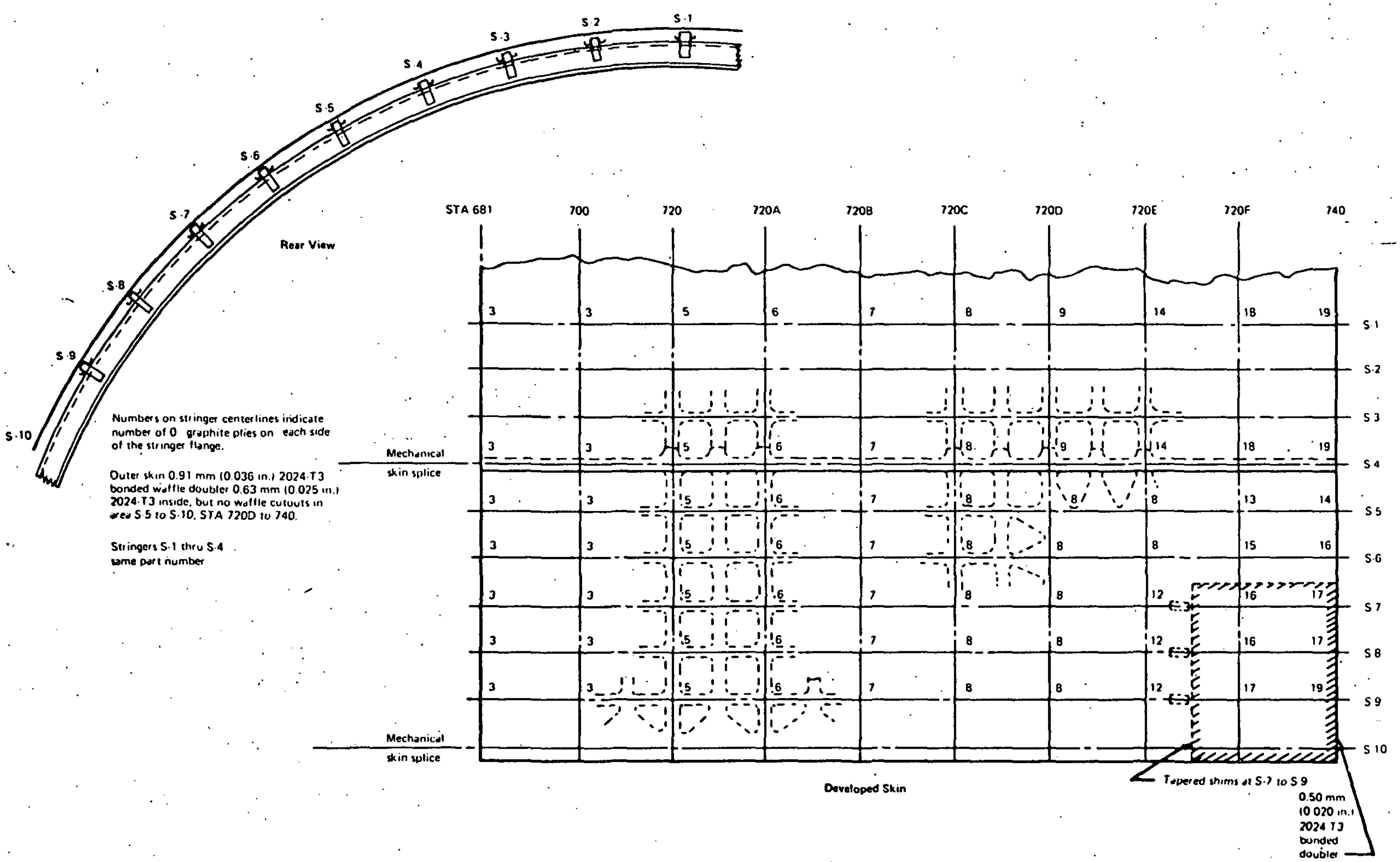


FIGURE 13.—CONCEPT 1—UPPER QUADRANT DESIGN

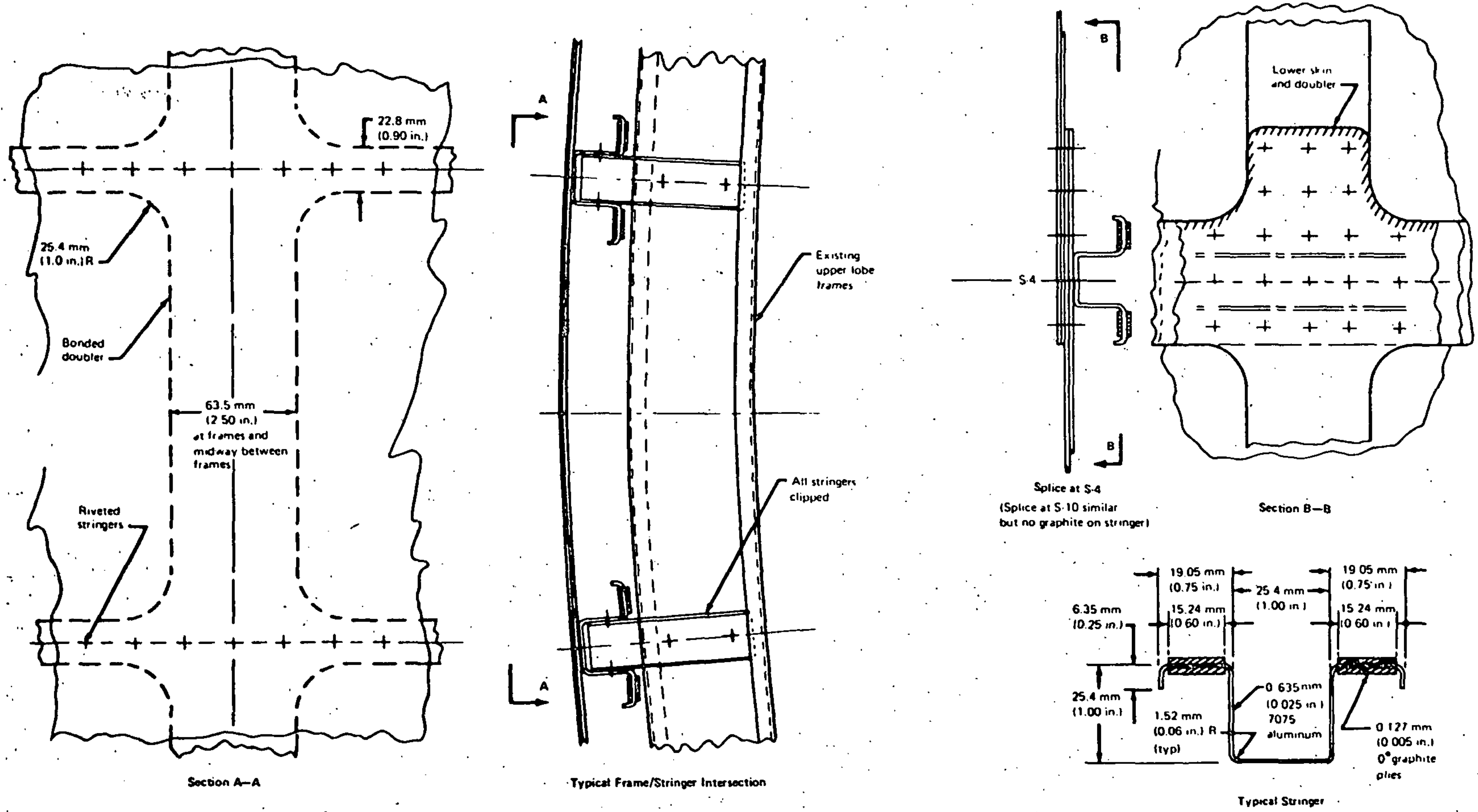


FIGURE 14. CONCEPT 1 - UPPER QUADRANT DESIGN DETAILS

FIGURE 14.—CONCEPT 1—UPPER QUADRANT DESIGN DETAILS

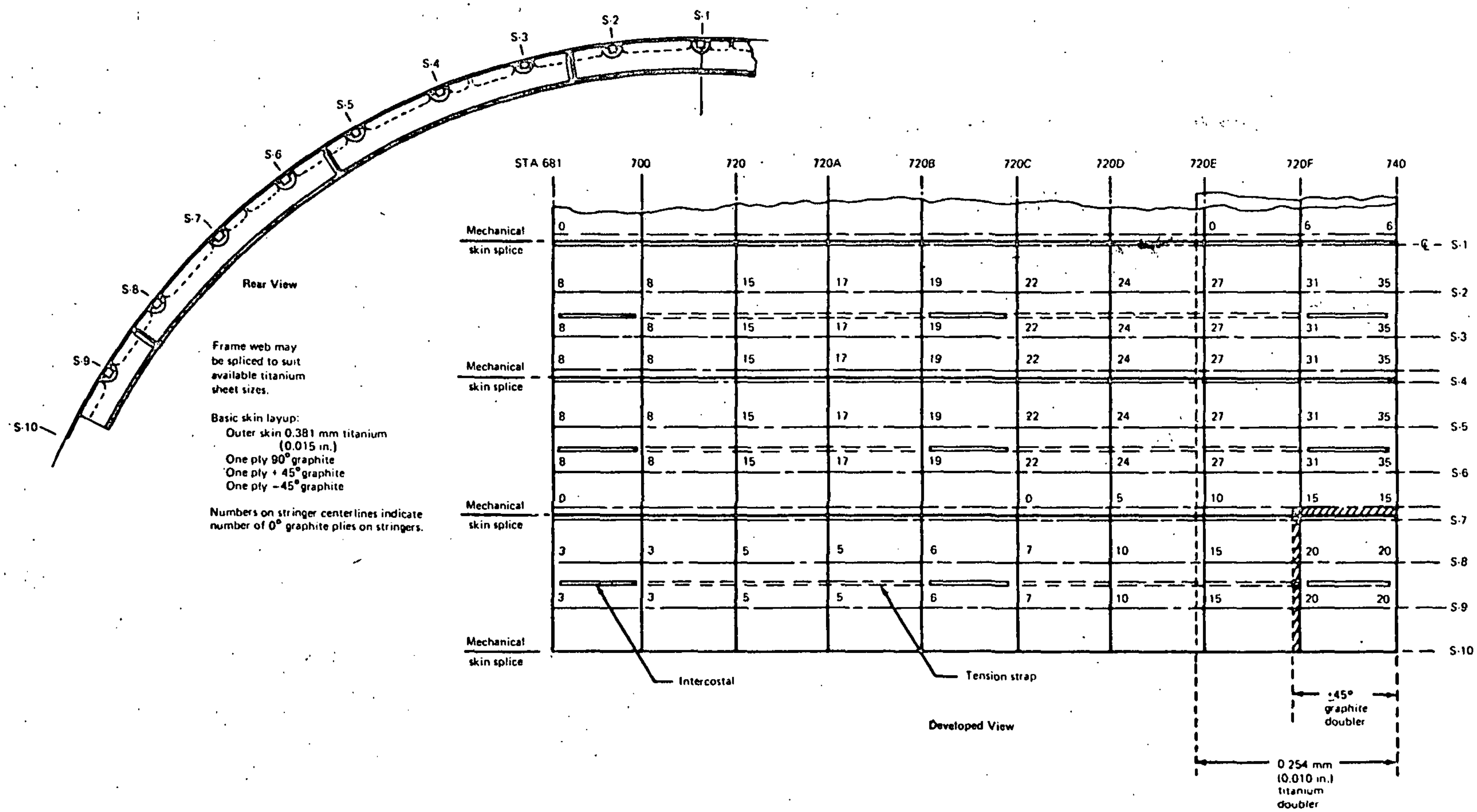


FIGURE 15.–CONCEPT 2–UPPER QUADRANT DESIGN

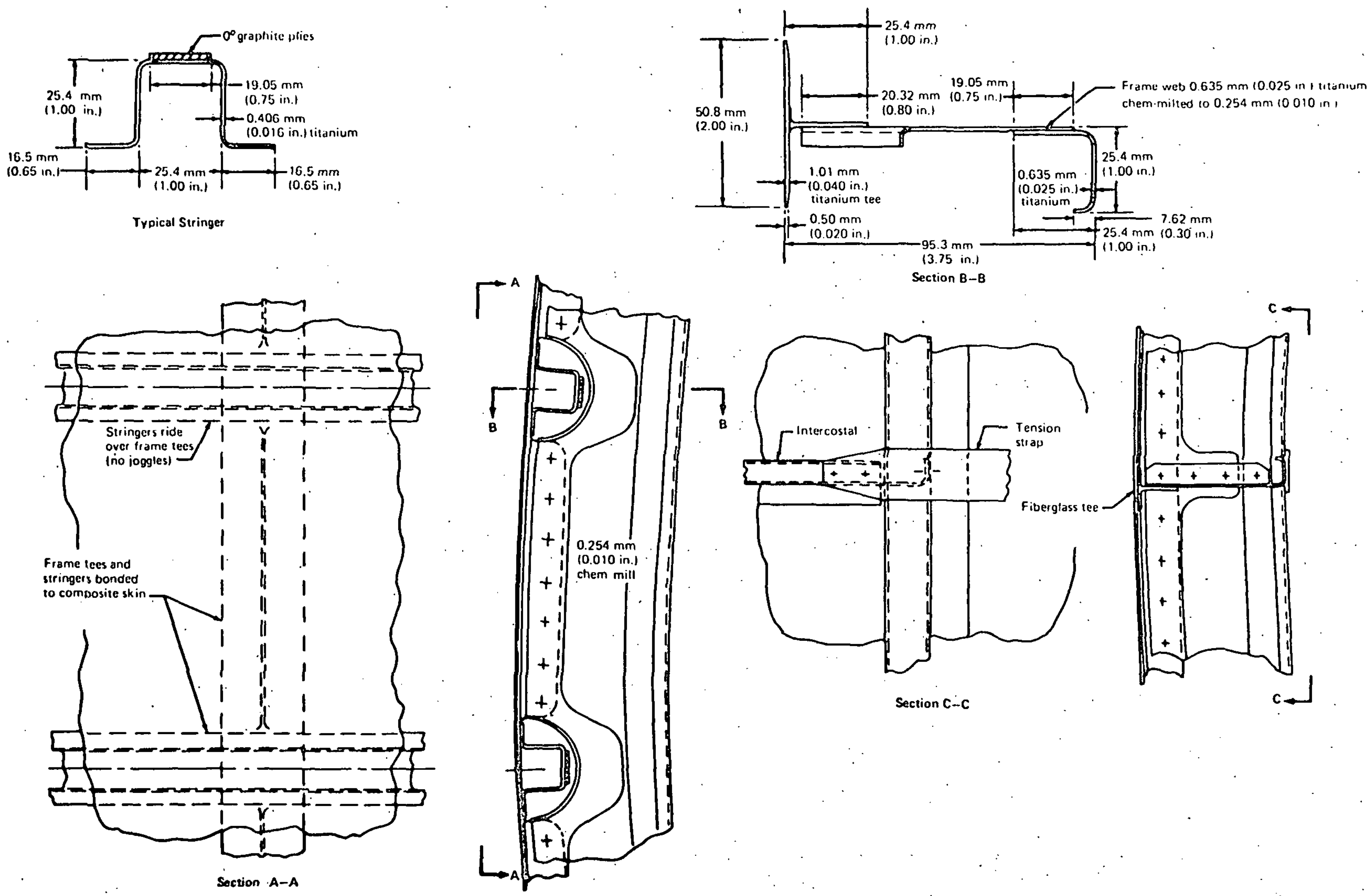


FIGURE 16.—CONCEPT 2—UPPER QUADRANT DESIGN DETAILS

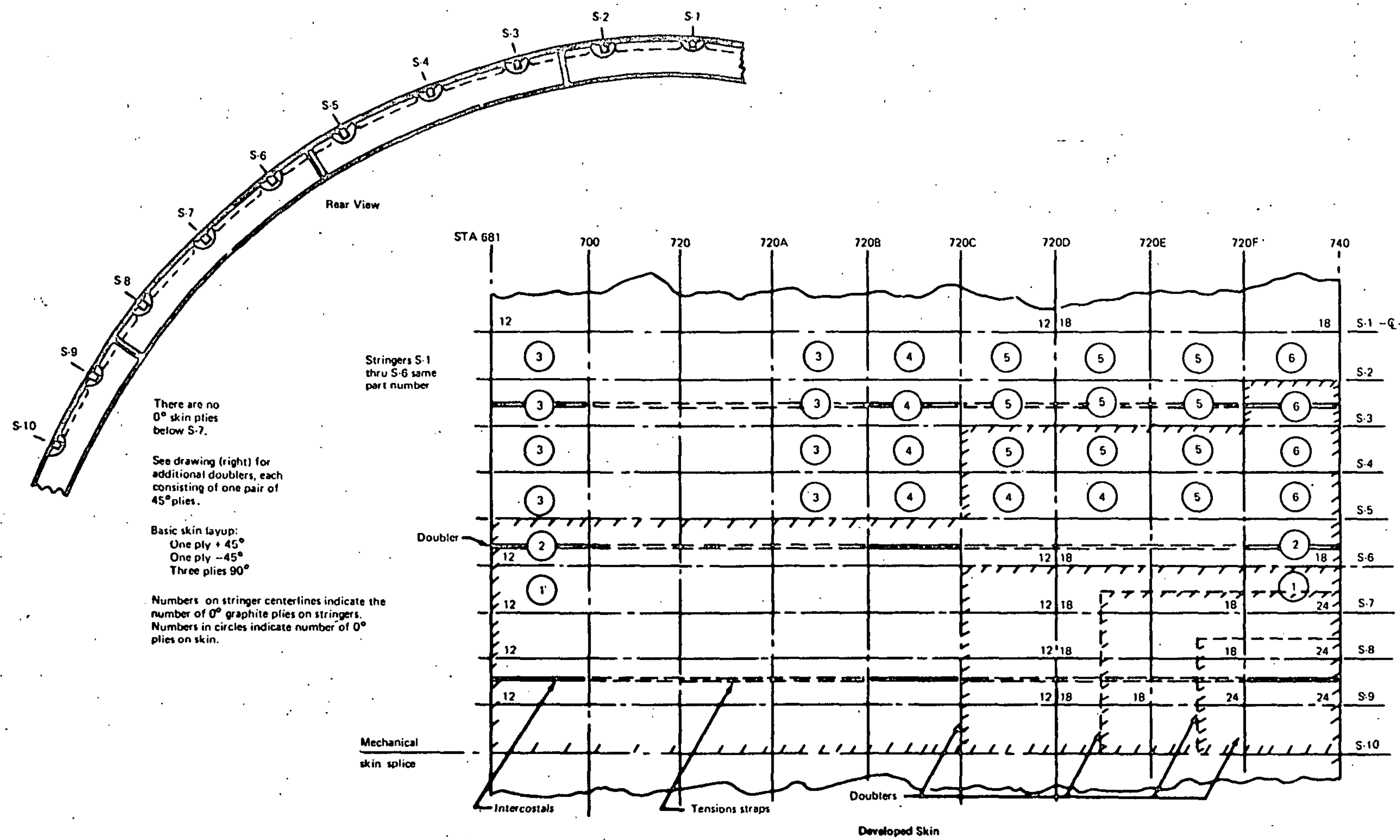


FIGURE 17.–CONCEPT 3–UPPER QUADRANT DESIGN

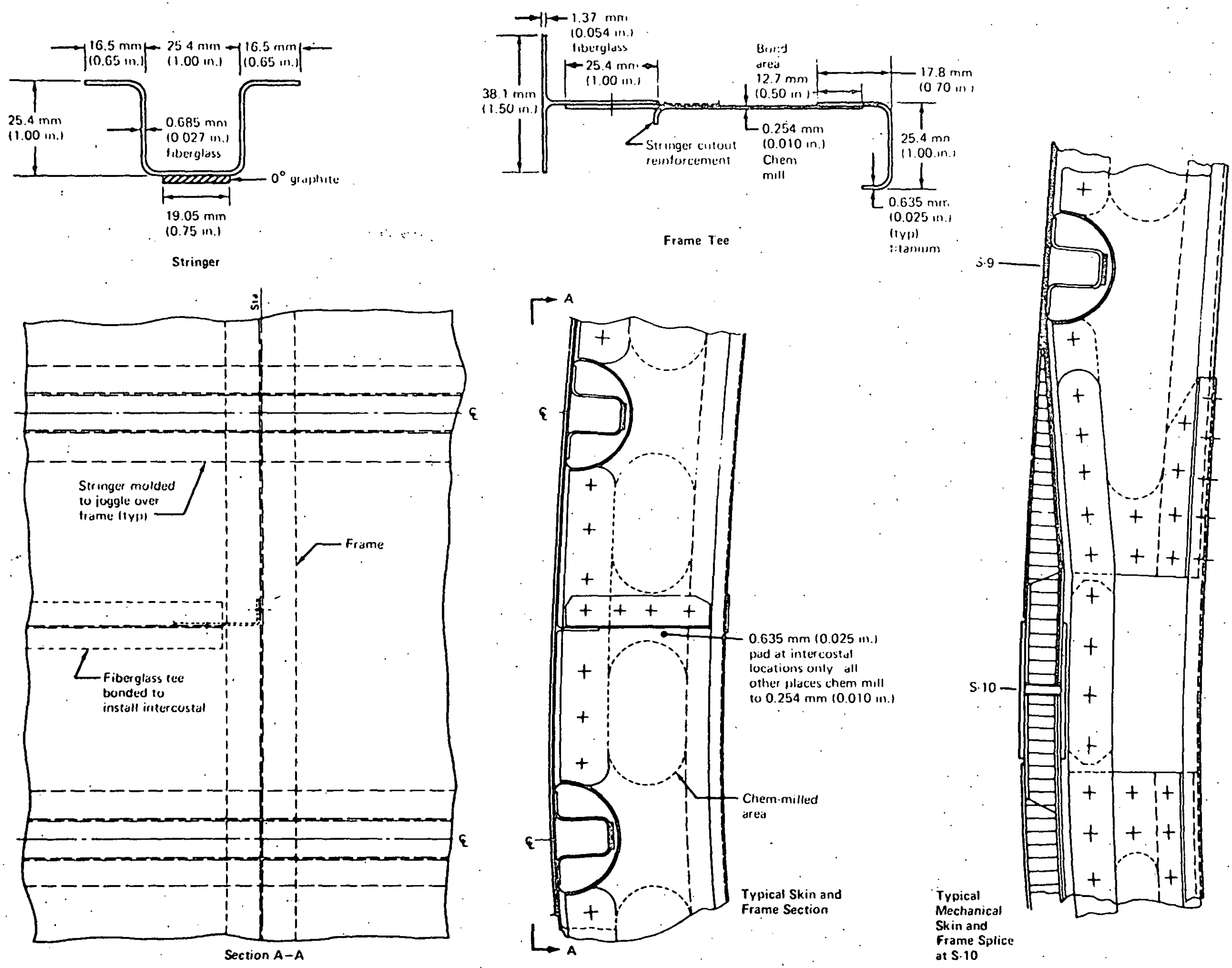


FIGURE 18.–CONCEPT 3–UPPER QUADRANT DESIGN DETAILS

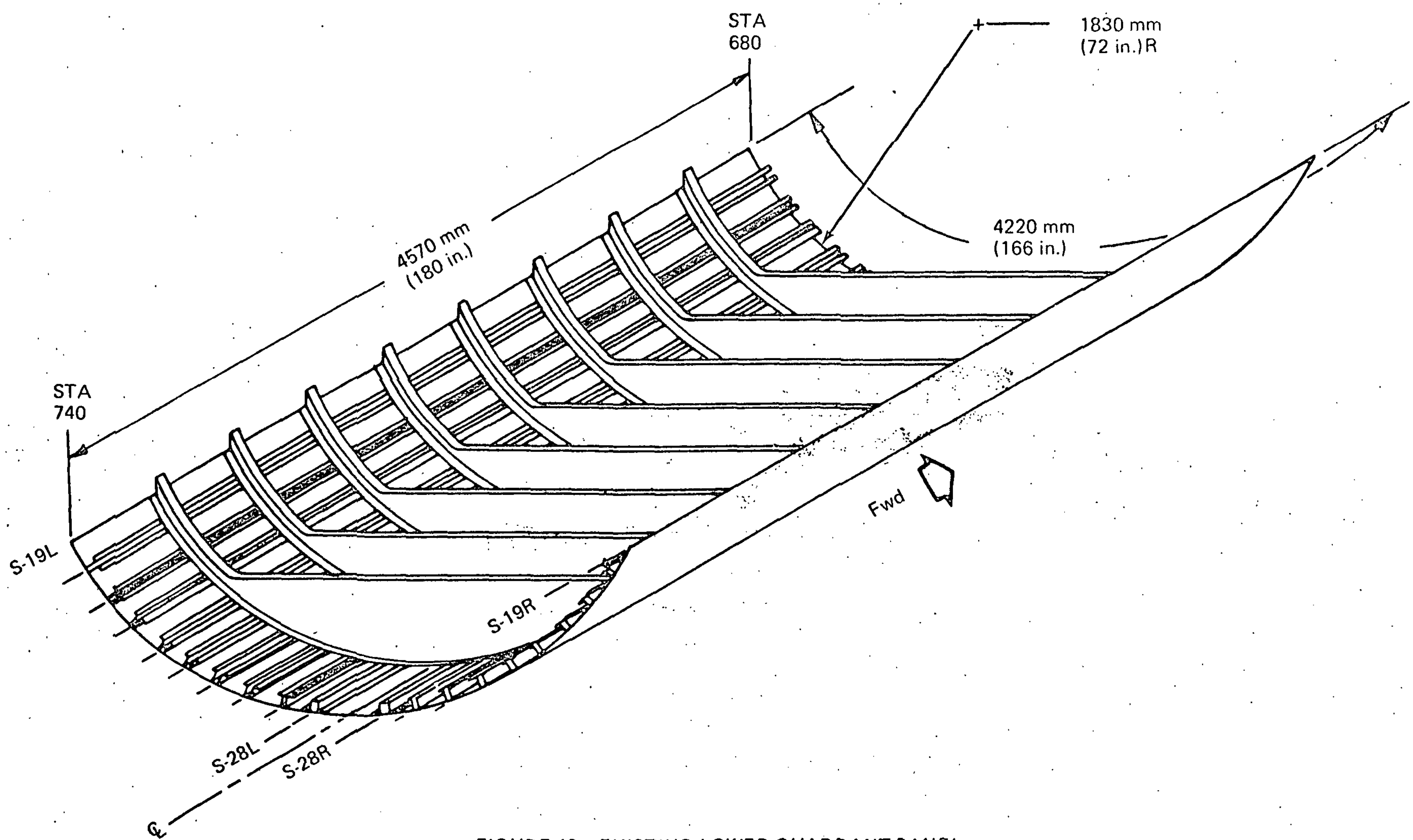


FIGURE 19.—EXISTING LOWER QUADRANT PANEL

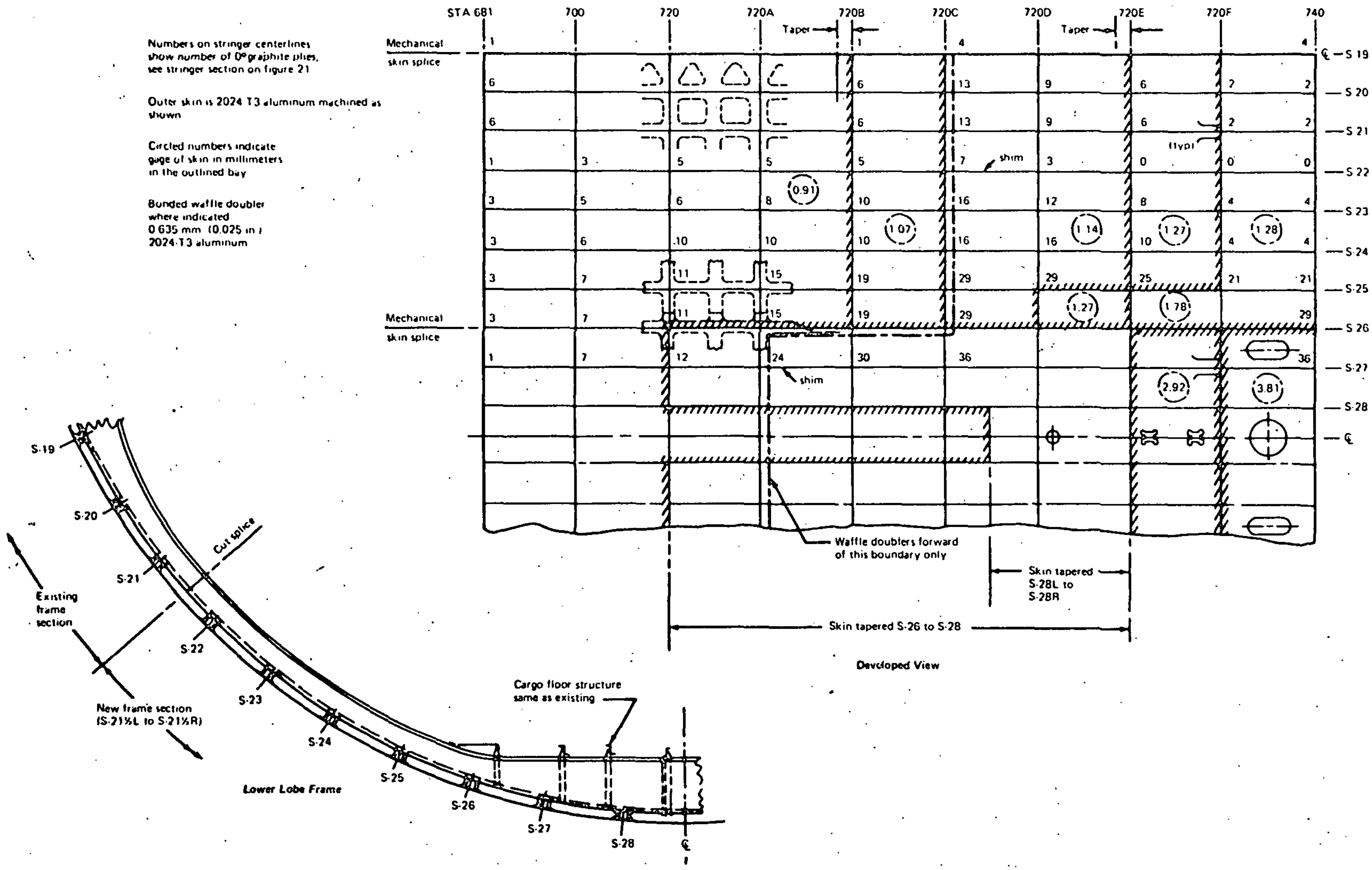


FIGURE 20.—CONCEPT 1—LOWER QUADRANT DESIGN

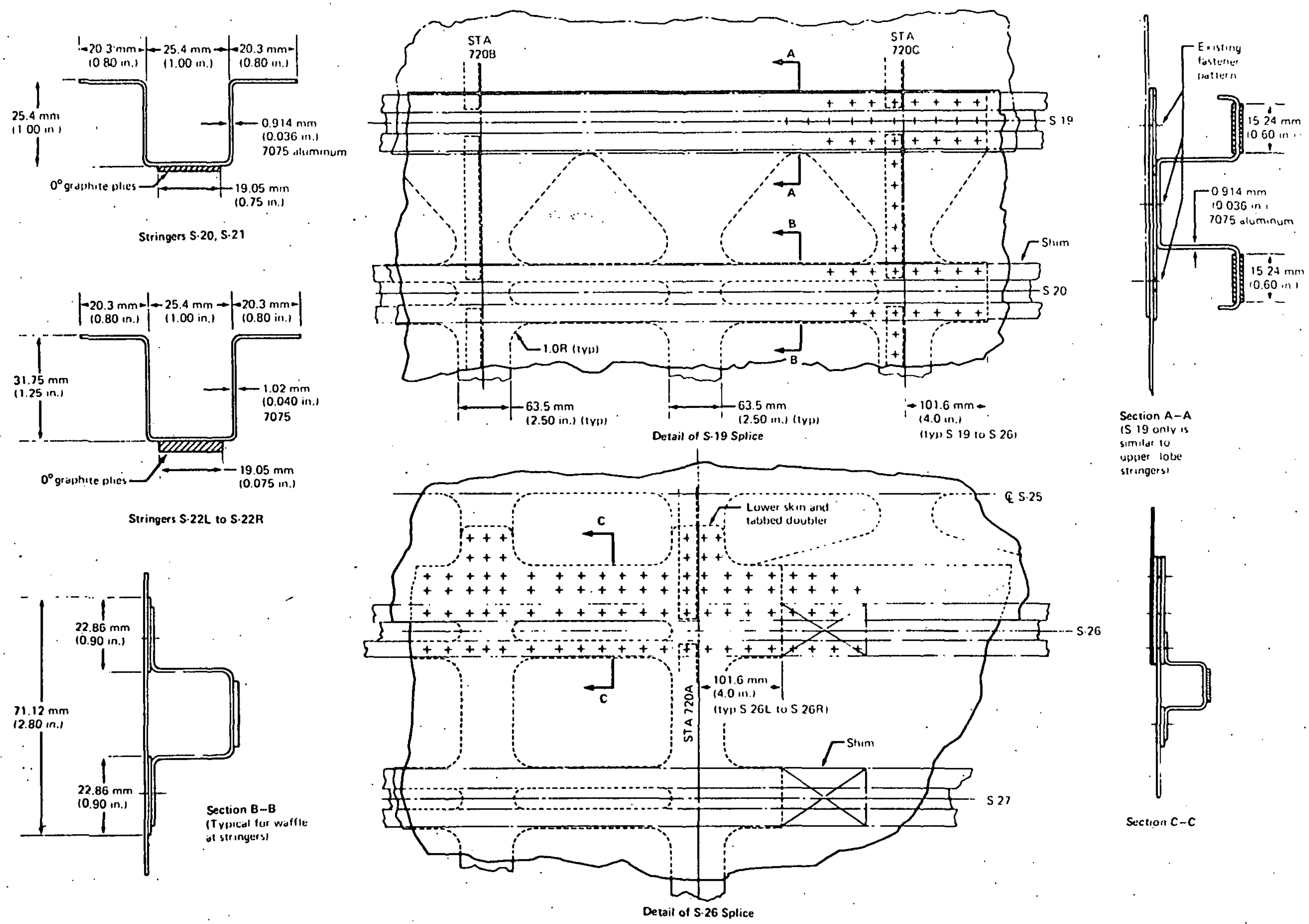


FIGURE 21.—CONCEPT 1—LOWER QUADRANT DESIGN DETAILS

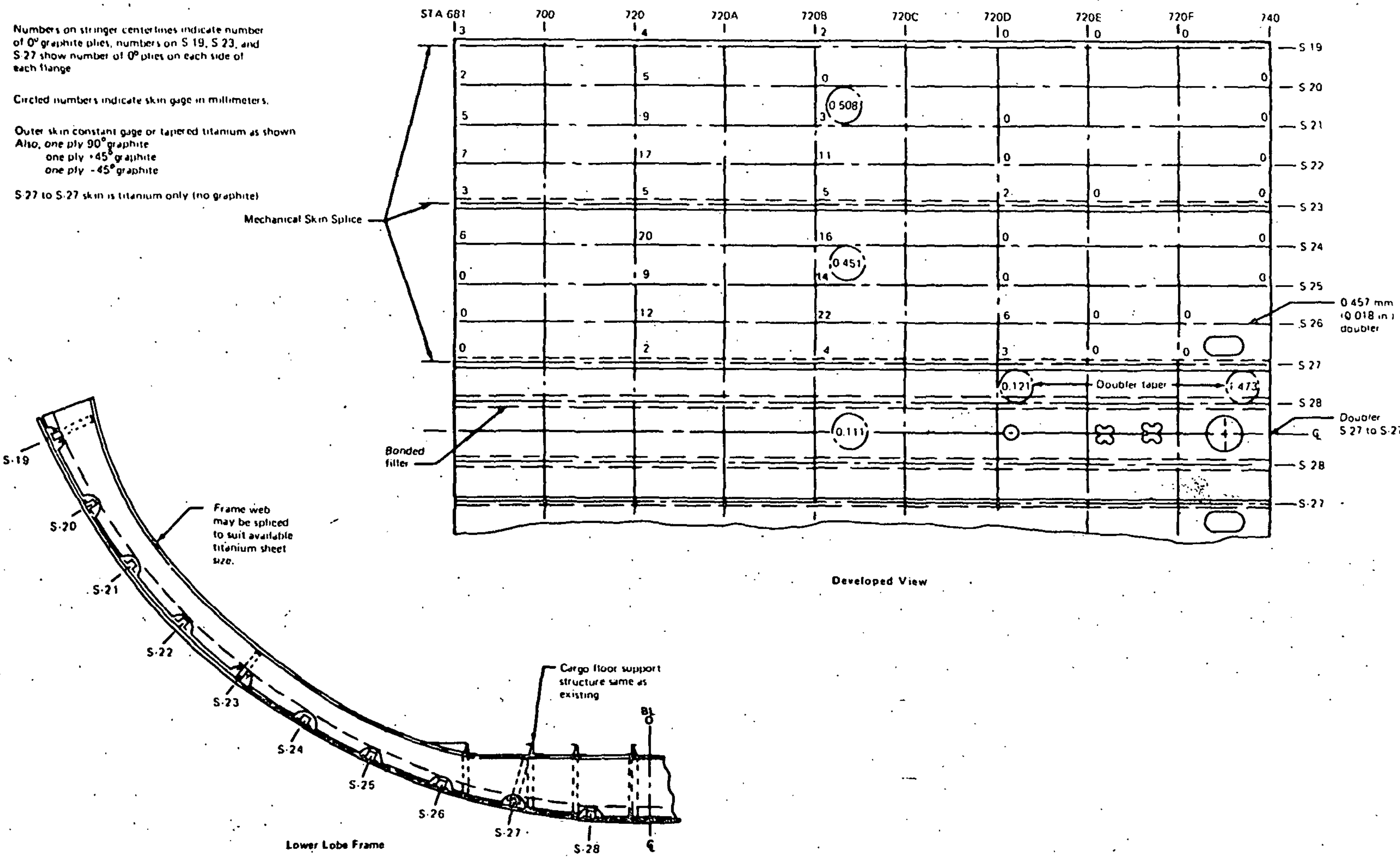


FIGURE 22.–CONCEPT 2–LOWER QUADRANT DESIGN

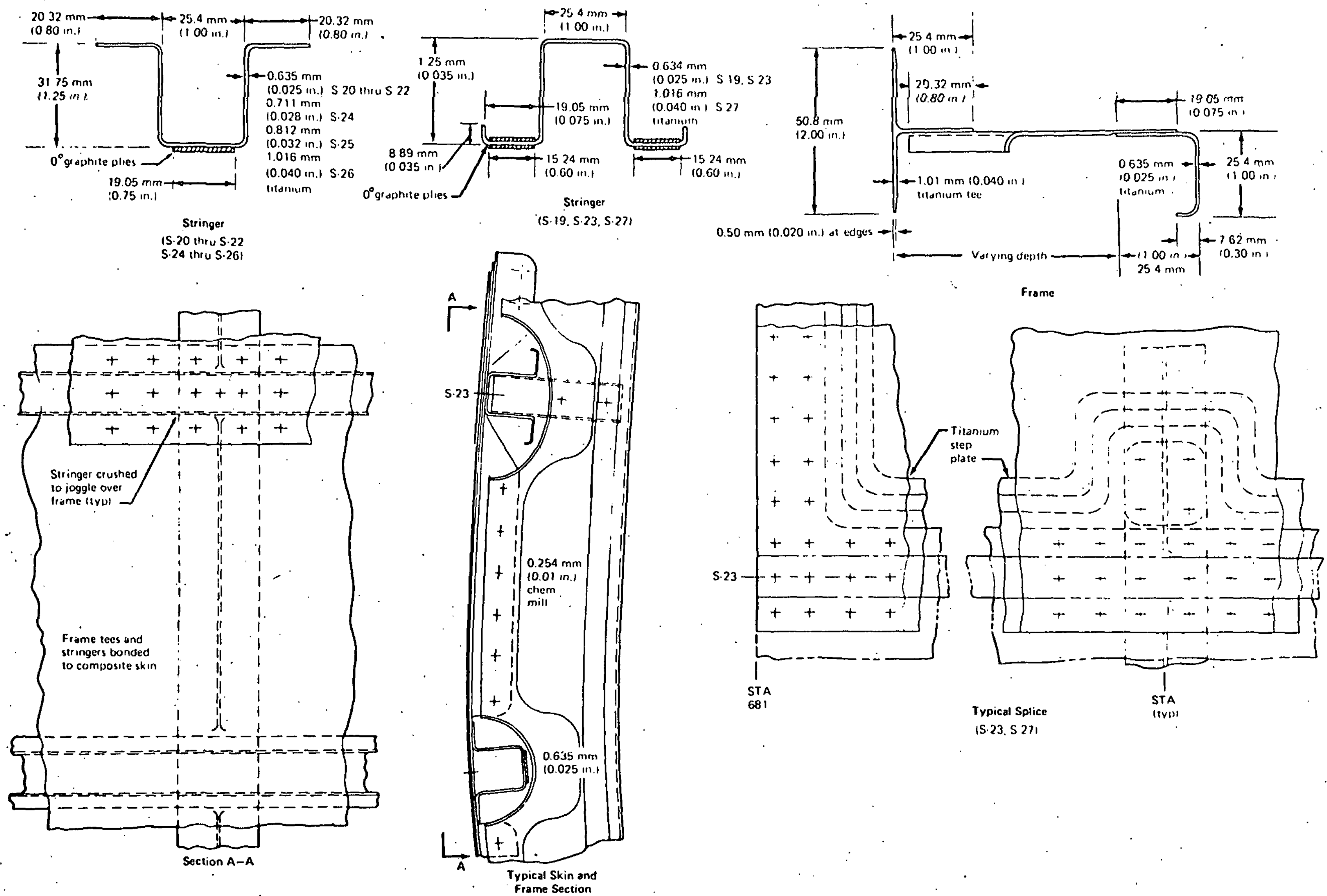


FIGURE 23.—CONCEPT 2—LOWER QUADRANT DESIGN DETAILS

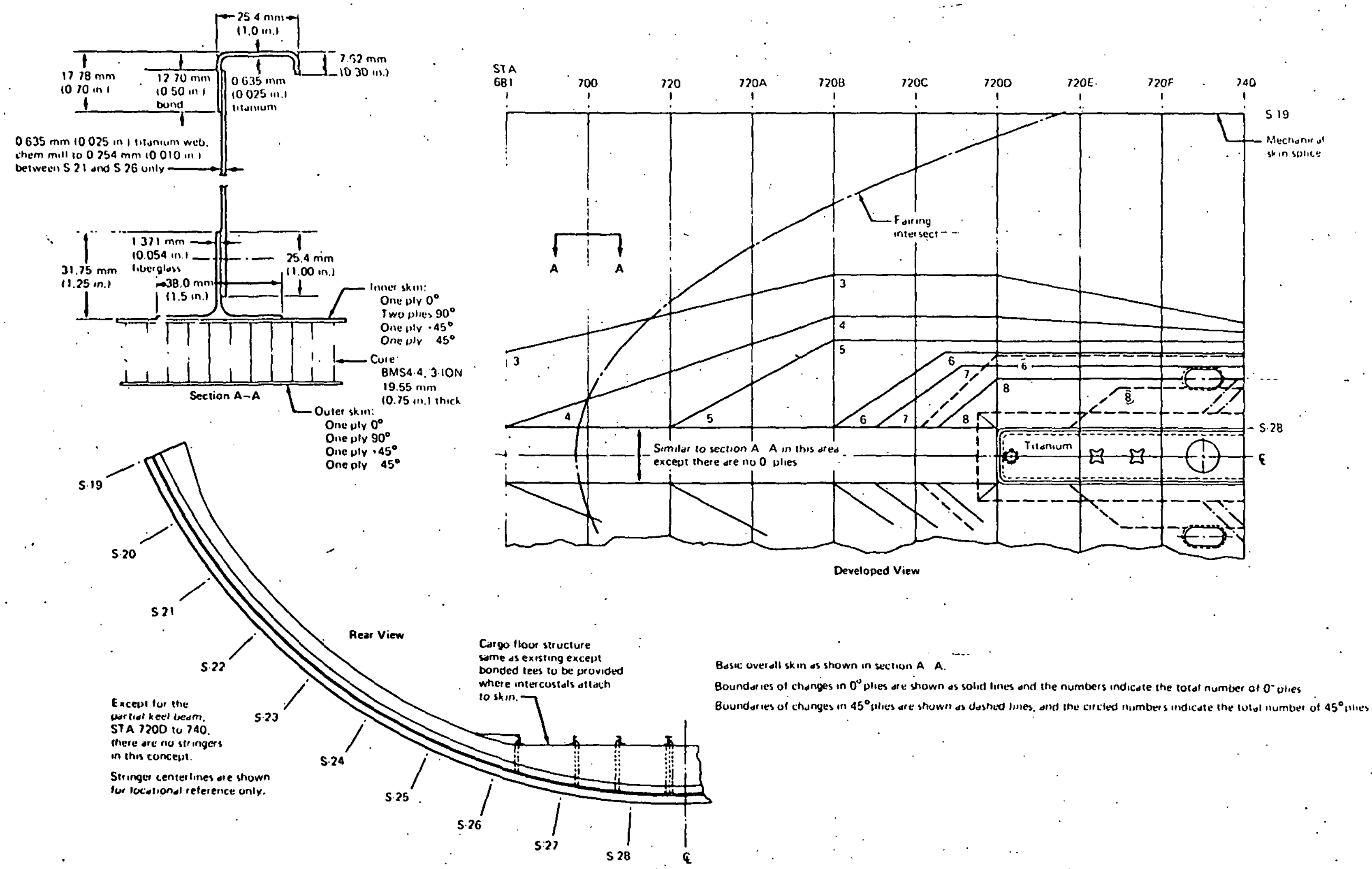


FIGURE 24.—CONCEPT 3—LOWER QUADRANT DESIGN

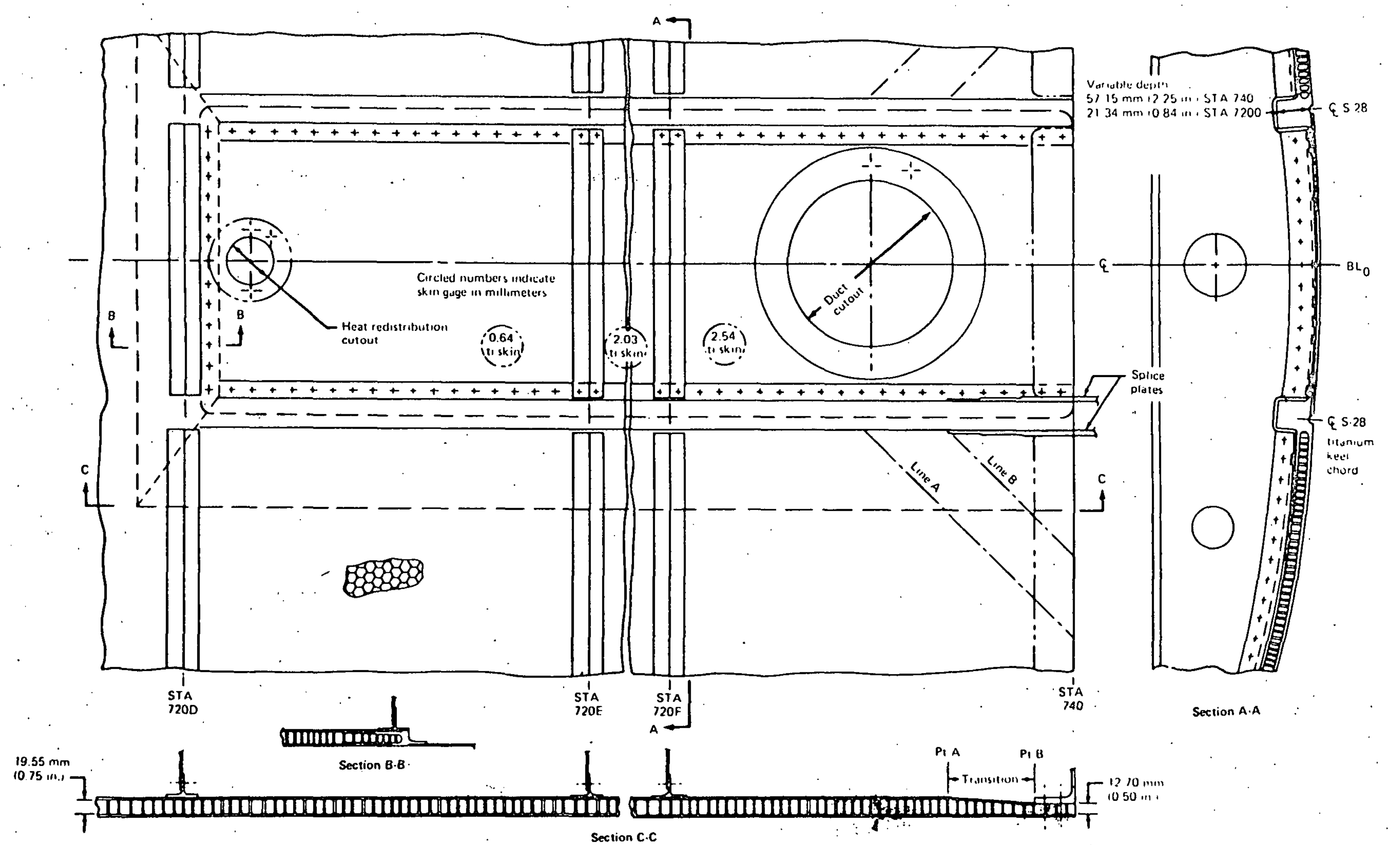


FIGURE 25.—CONCEPT 3—LOWER QUADRANT DESIGN DETAILS

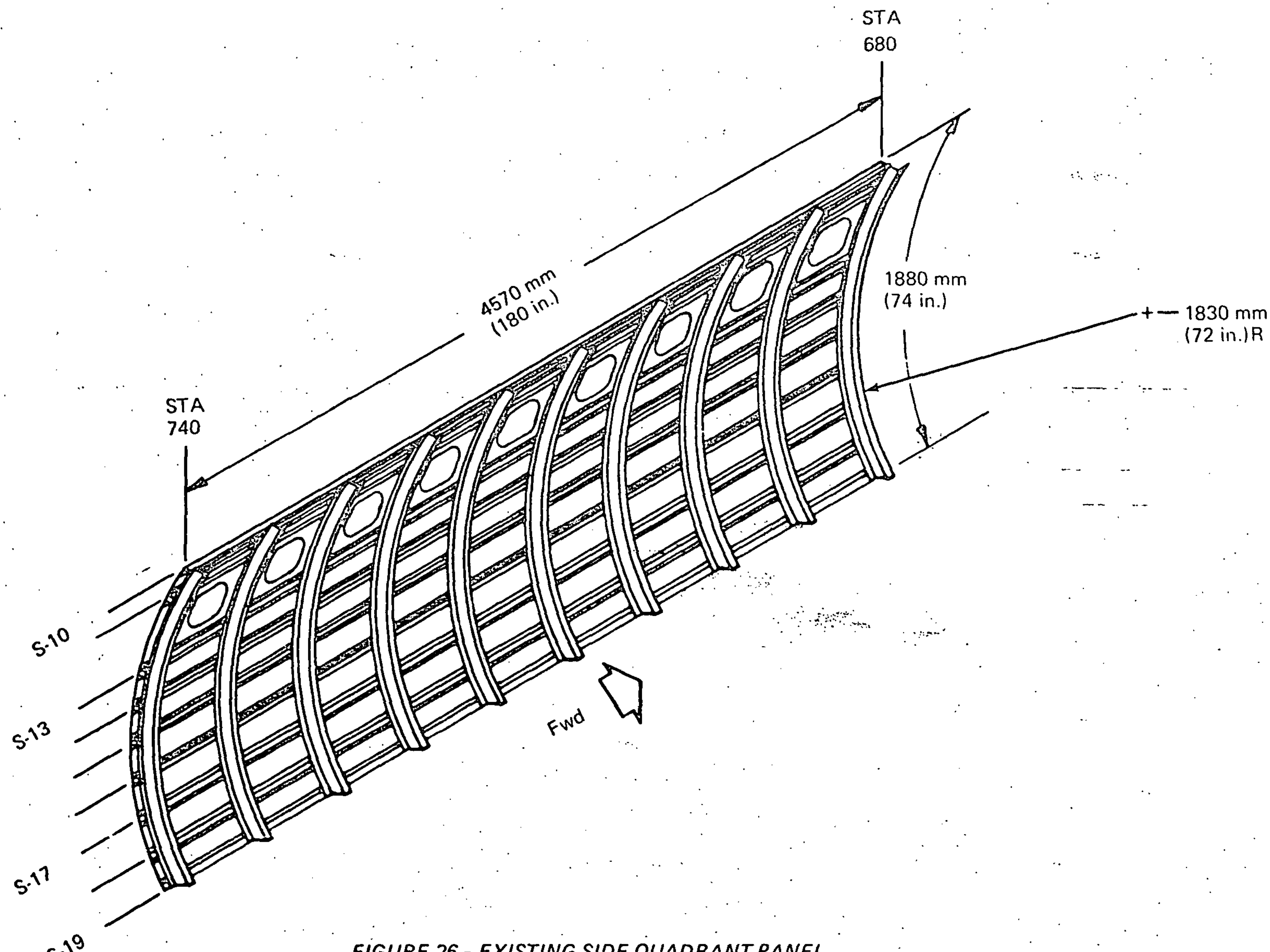


FIGURE 26.—EXISTING SIDE QUADRANT PANEL

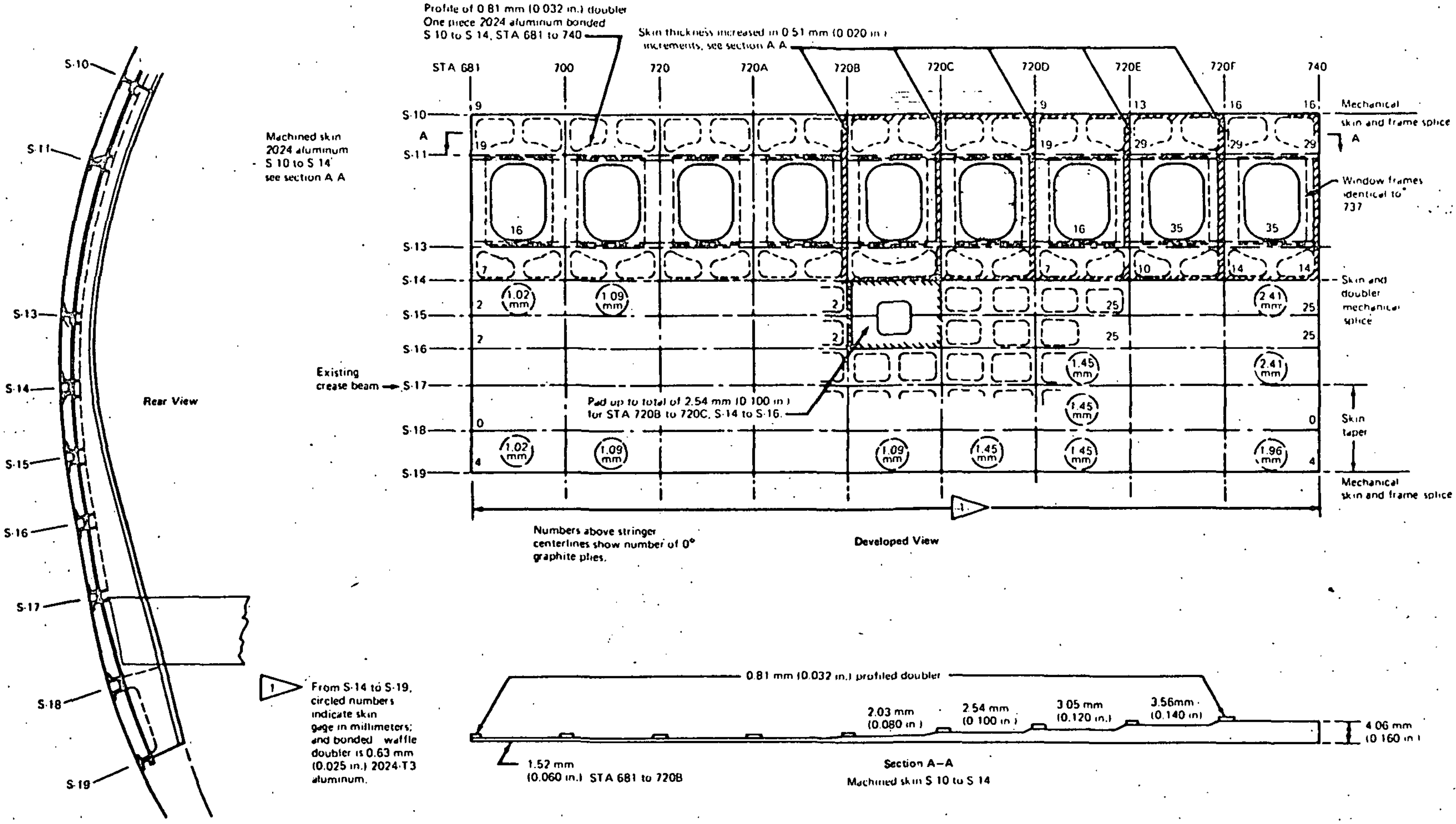


FIGURE 27.—CONCEPT 1—SIDE QUADRANT DESIGN

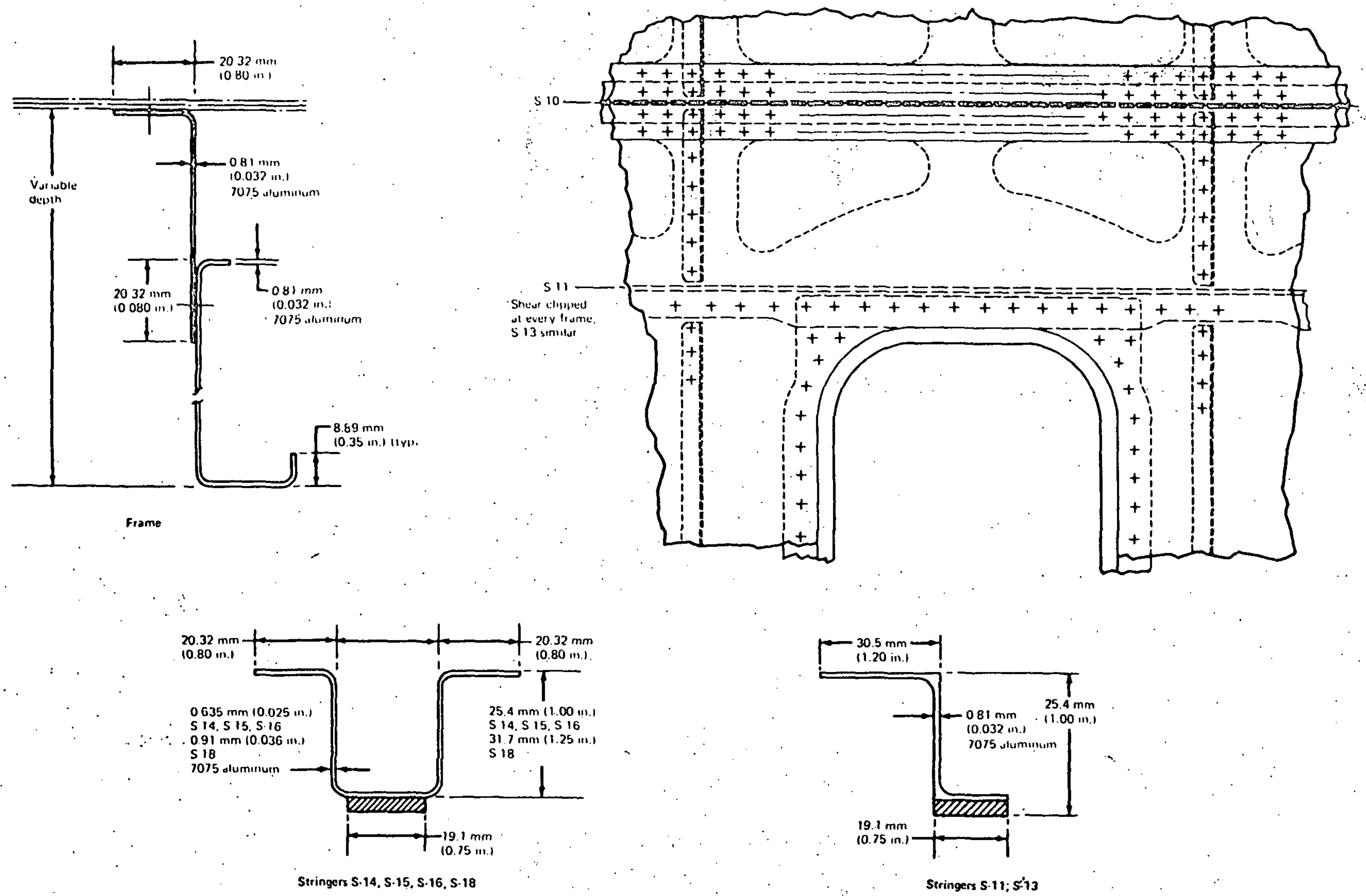


FIGURE 28.—CONCEPT 1—SIDE QUADRANT DESIGN DETAILS

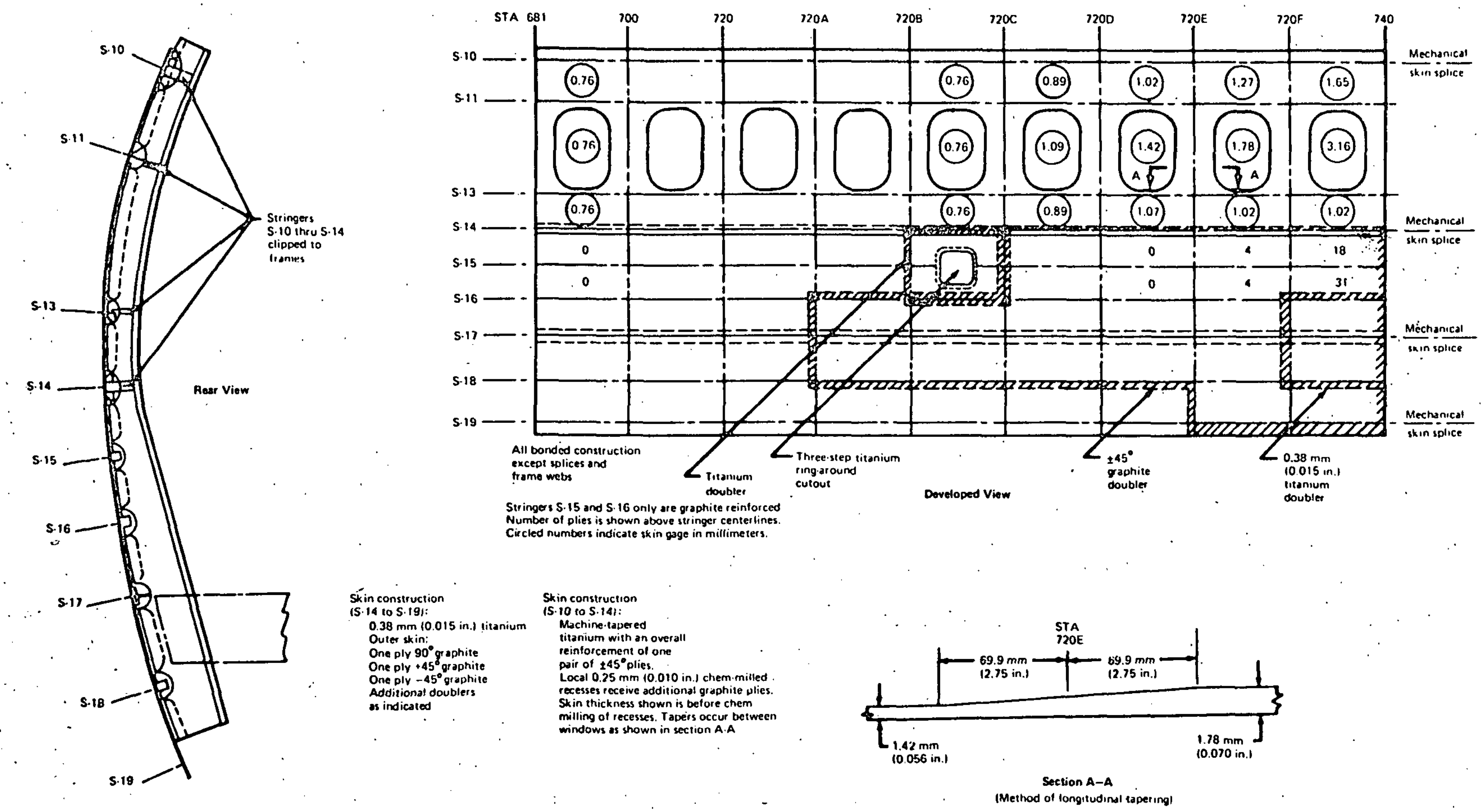


FIGURE 29.–CONCEPT 2–SIDE QUADRANT DESIGN

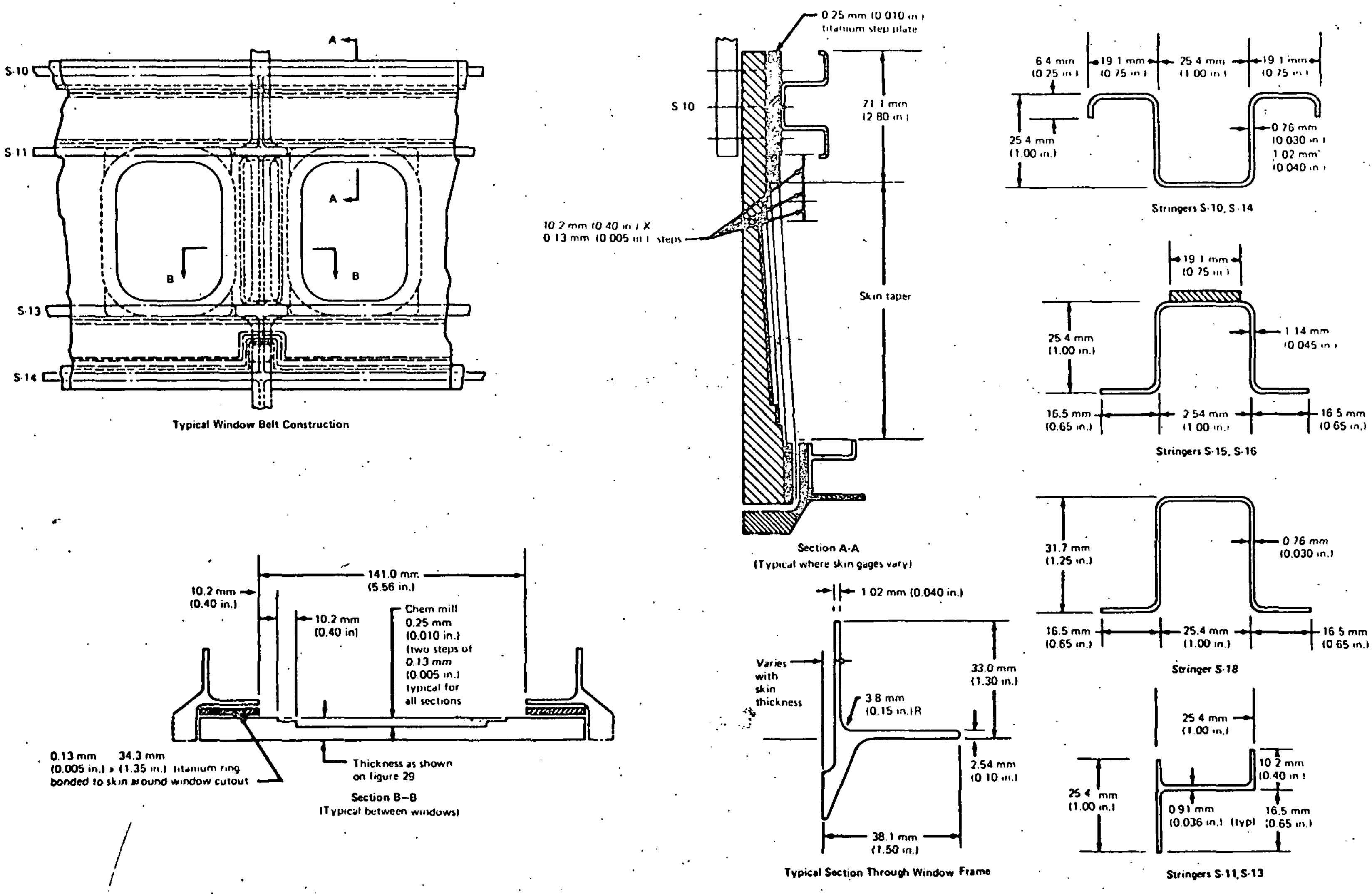


FIGURE 30.–CONCEPT 2–SIDE QUADRANT DESIGN DETAILS

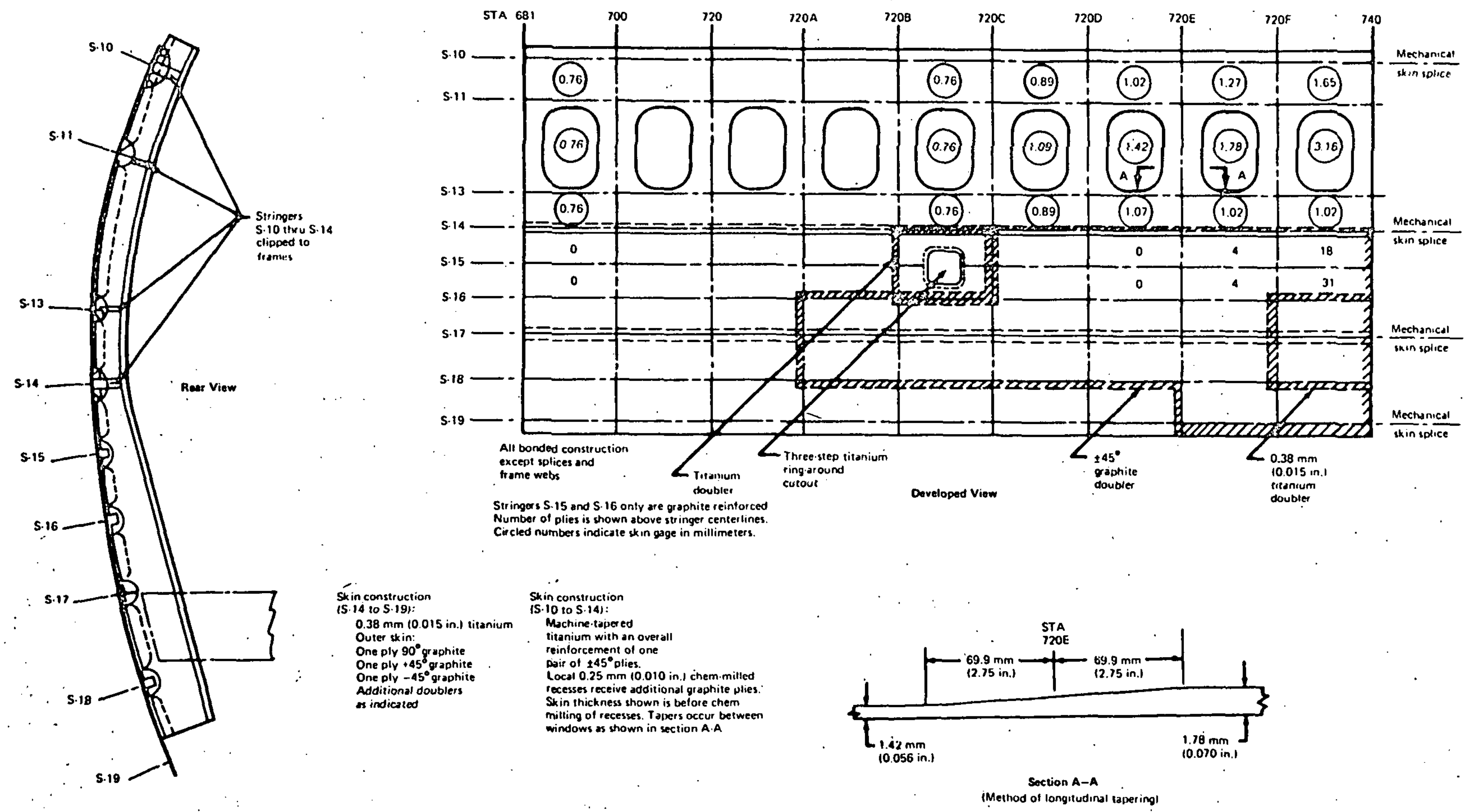


FIGURE 29.—CONCEPT 2—SIDE QUADRANT DESIGN

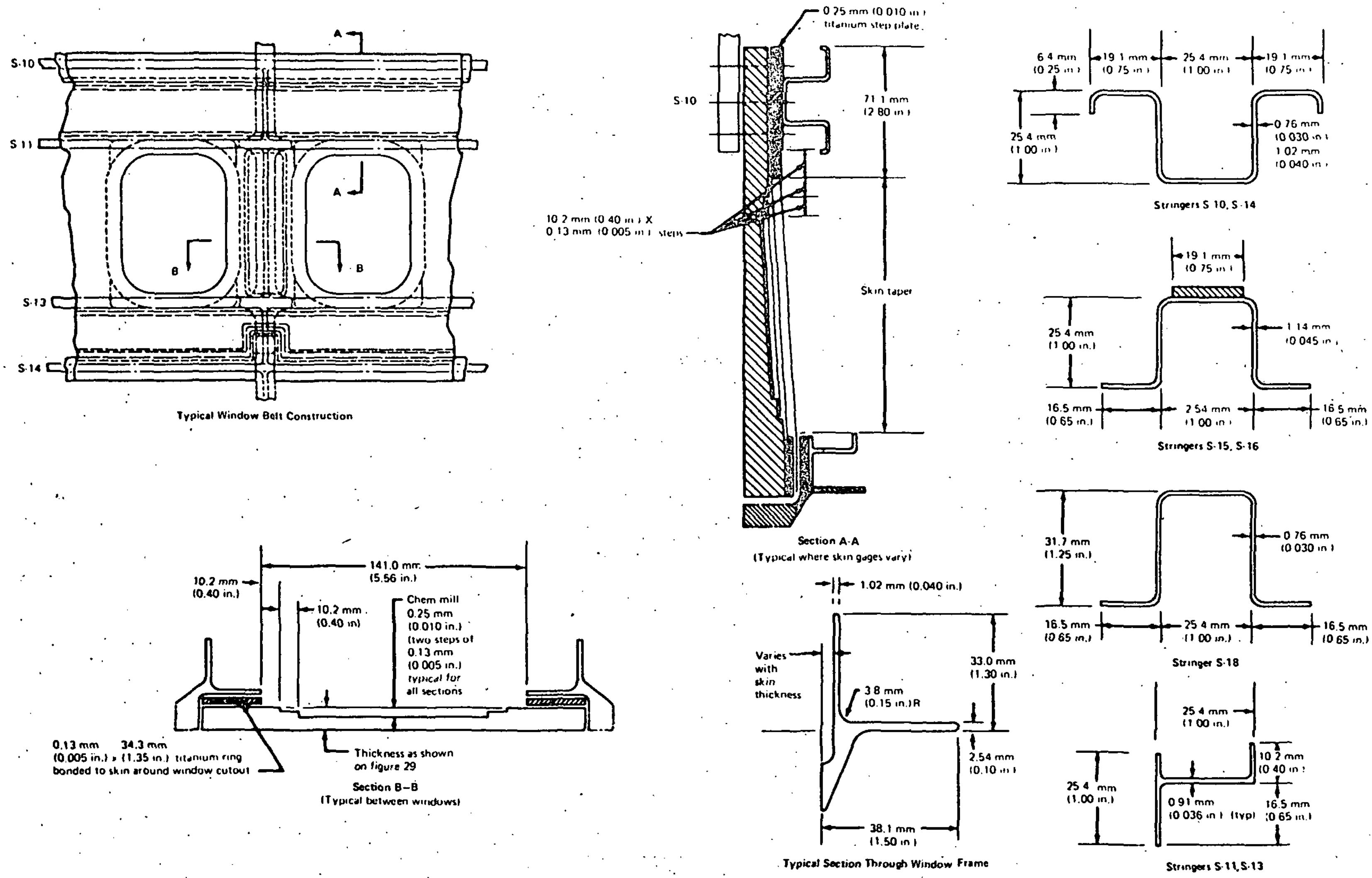


FIGURE 30.–CONCEPT 2–SIDE QUADRANT DESIGN DETAILS

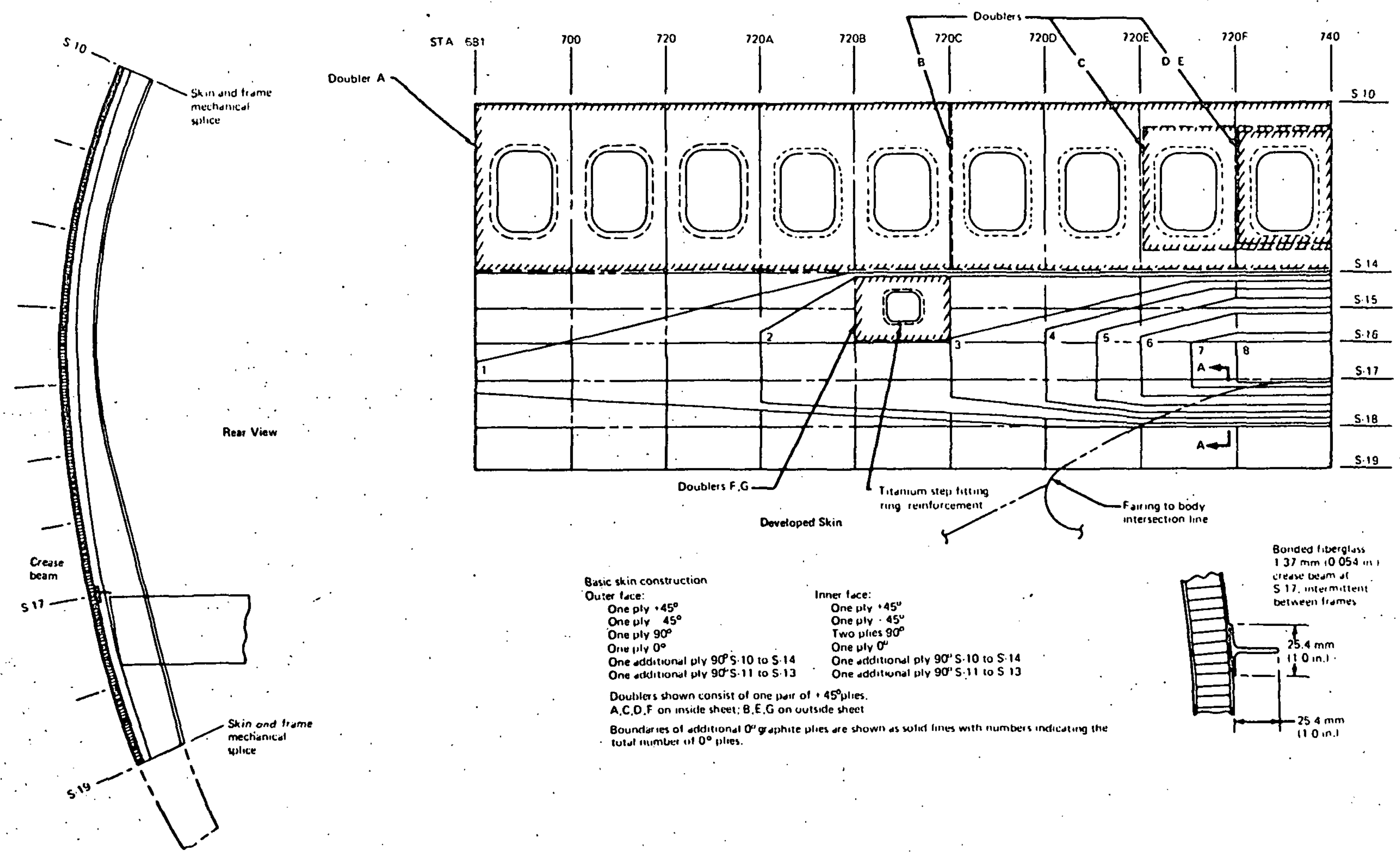


FIGURE 31.—CONCEPT 3—SIDE QUADRANT DESIGN

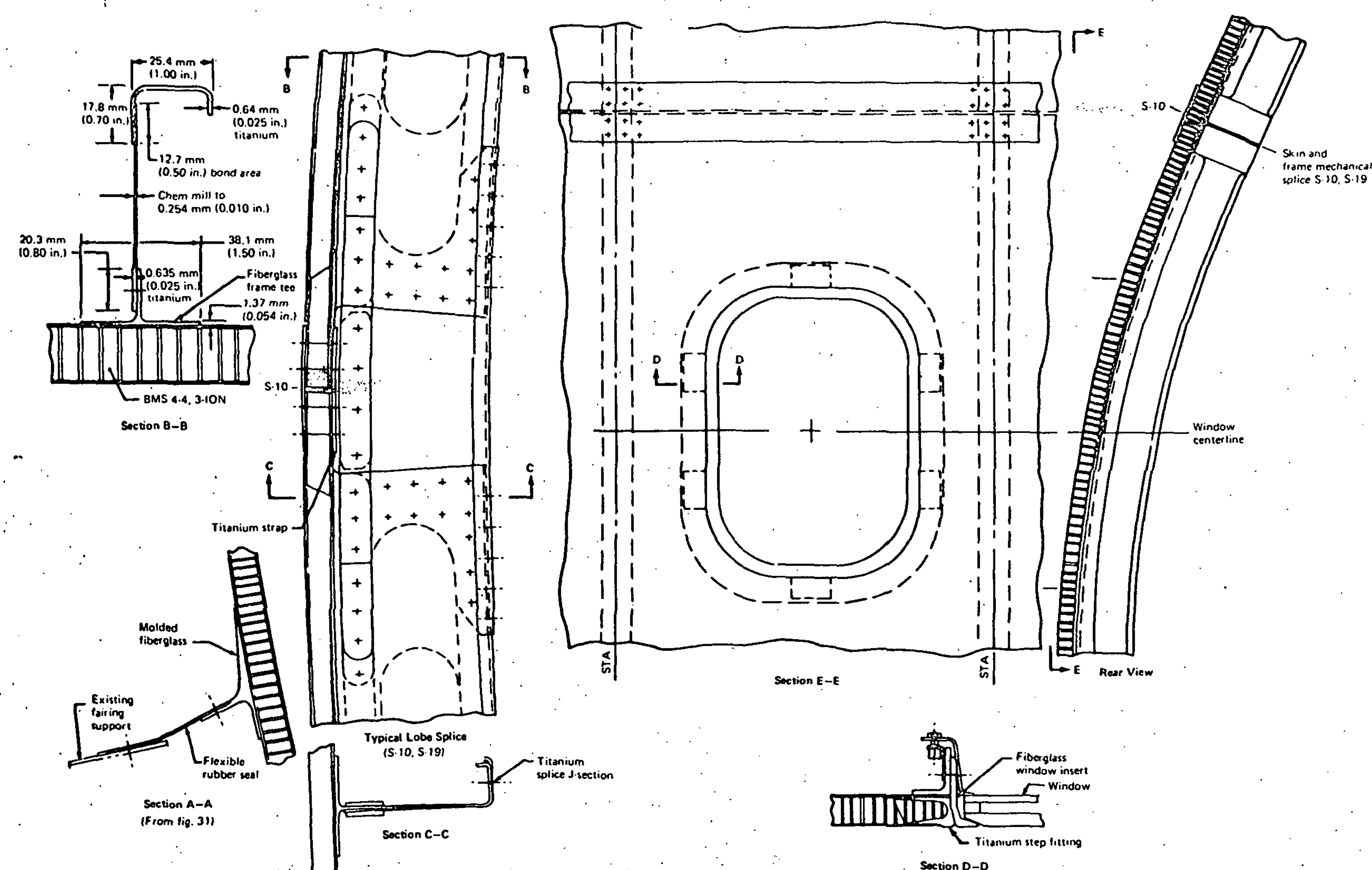


FIGURE 32.–CONCEPT 3–SIDE QUADRANT DESIGN DETAILS

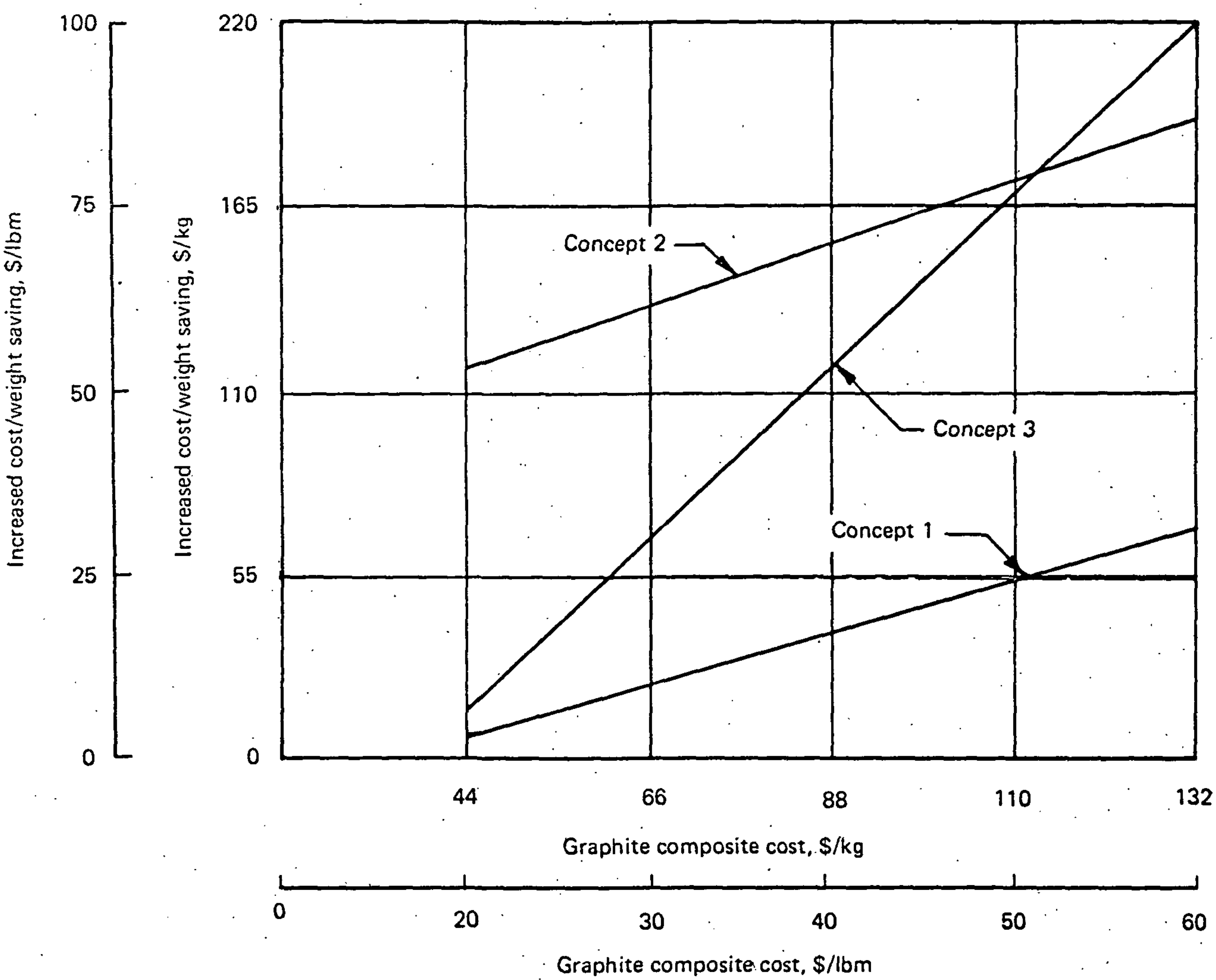


FIGURE 33.—EFFECT OF GRAPHITE COMPOSITE PRICE ON THE COST OF WEIGHT SAVING

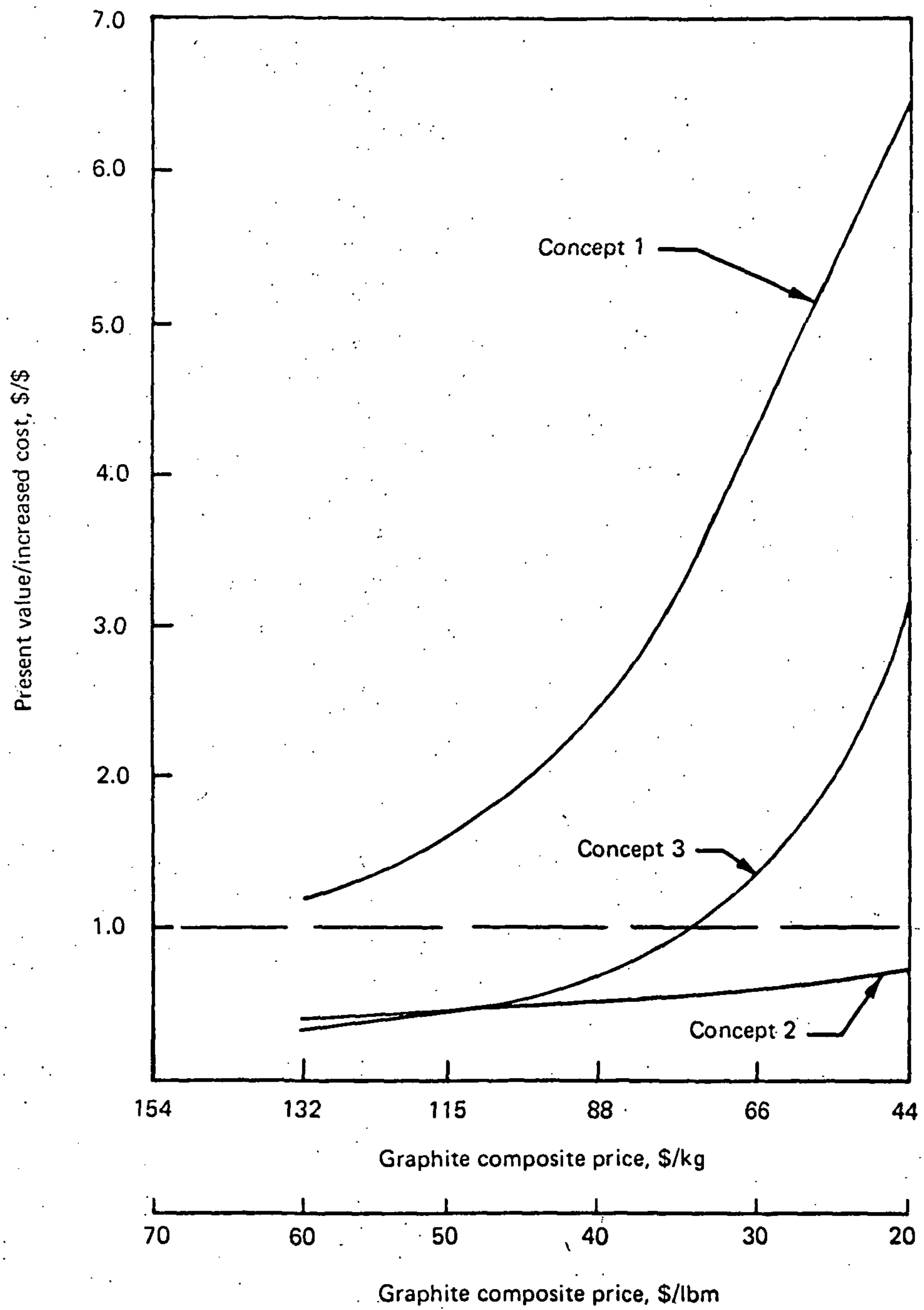


FIGURE 34.—RATIO OF PRESENT VALUE TO INCREASED PRODUCTION COST VS COST OF GRAPHITE COMPOSITE

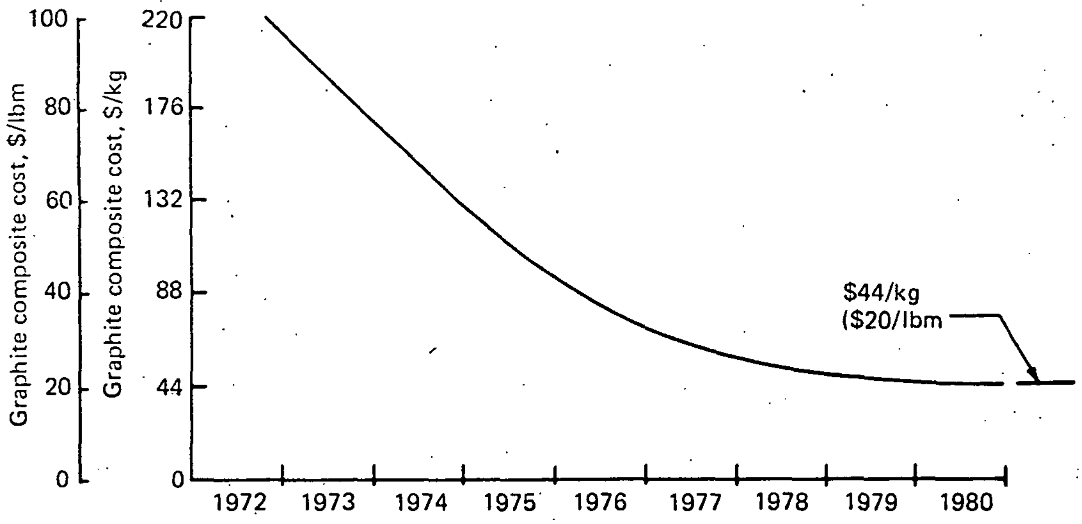


FIGURE 35.—PROJECTED COST FOR HIGH-STRENGTH GRAPHITE PREPREG

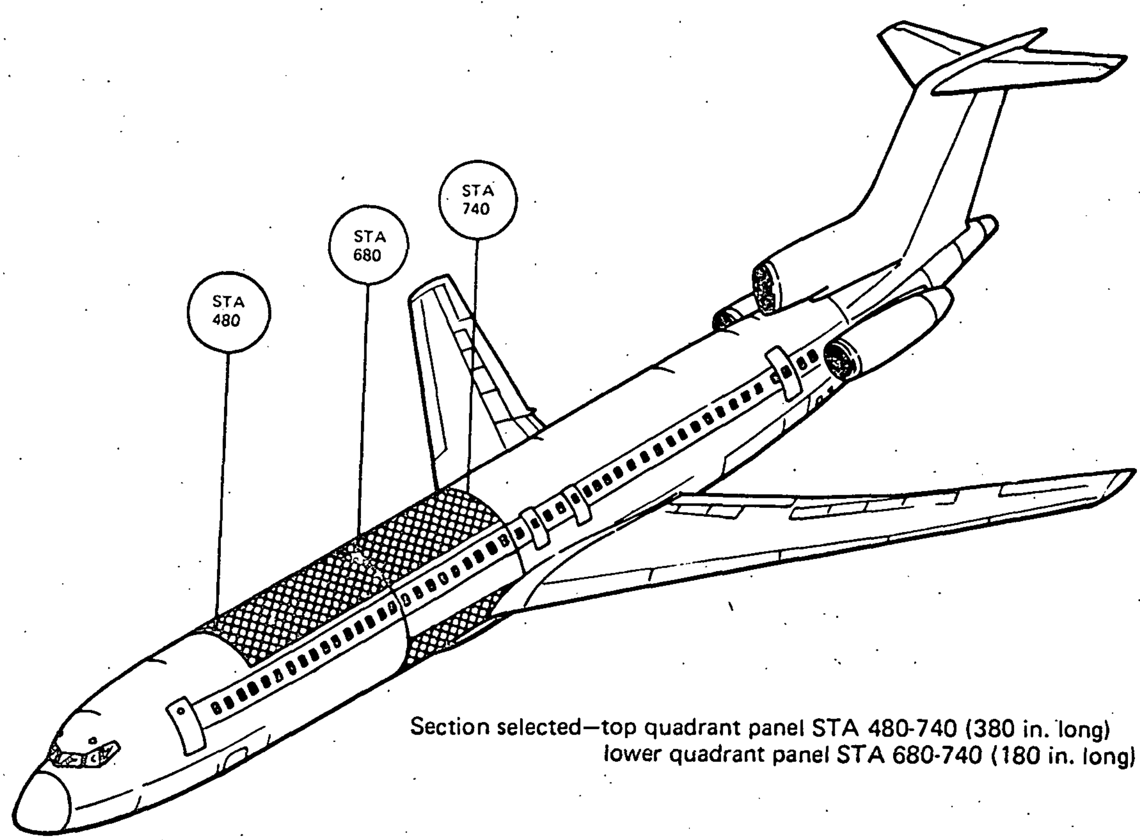


FIGURE 36.—CONCEPT 1—FLIGHT SERVICE EVALUATION AIRCRAFT

4.0 DEMONSTRATION PROGRAM PLAN

The demonstration program plan for all three concepts is divided into five phases, as follows:

- Phase I—developmental program
- Phase II—design, analysis, and engineering verification tests
- Phase III—fabrication and quality assurance
- Phase IV—full-scale ground tests and documentation
- Phase V—flight service evaluation

A discussion of each phase is contained in the following sections. The schedules for all five phases for concepts 1, 2, and 3 are shown in figures 37, 38, and 39, respectively. Each of the following sections contains a general program discussion which relates to all three concepts. Specific tests that relate to the concept 1, 2, and 3 designs are contained in appendixes A, B, and C, respectively.

4.1 PHASE I—DEVELOPMENTAL PROGRAM

The developmental program will provide the basic engineering knowledge necessary to proceed with the final design of the components. The major portion of this phase will be completed before final release of the component drawings. The adhesive environmental exposure tests will not be completed before final drawing release; however, these data will be available before the start of the flight service test.

4.1.1 Composite Laminate Selection

The composite laminates defined in table 2 will be evaluated, and the most suitable laminate will be selected on the basis of repeatability of mechanical properties, bond strength to the base metal, and handling characteristics for fabrication processes. A procurement specification will be developed as a result of this series of tests. The test program for each concept is contained in the appendixes.

4.1.2 Adhesive Selection, Laminate Bonding, and Titanium Surface Treatment

The candidate adhesive systems defined in table 3 will be evaluated. AF 30 is presently considered to be the only 450° K (350° F) system that will be used; therefore, no selection process will be needed. The selection of the 394° K (250° F) system will be based on evaluation according to criteria contained in Boeing materials specifications. Standard 12.7-mm (0.5-in.) overlap specimens will be used in room temperature tests and environmental

resistance will be determined using stressed specimens exposed to elevated temperature and humidity. Along with this evaluation of the 394° K (250° F) systems on metal substrates, the compatibility of these systems with the selected composite will be evaluated. An adhesive material specification and a process specification will be written.

A titanium surface treatment program will be conducted in conjunction with the composite bonding program. This program will evaluate the compatibility of present surface treatment methods with the selected composite and adhesive systems. Environmental tests will be conducted on laboratory specimens, and the titanium surface treatment process will be evaluated.

The composite bonding and surface treatment programs will be the same for all concepts.

4.1.3 Adhesive Laboratory and Outdoor Exposure Tests

Laboratory and outdoor exposure tests will be conducted using AF 30 and the selected 394° K (250° F) system. Since laboratory tests indicate that adhesive bond delamination increases with increasing humidity and temperature, a high-humidity, warm climate such as that of Panama will be selected for the outdoor exposure tests. The laboratory exposures will simulate critical ground and flight service conditions by subjecting test specimens to varying temperature and humidity levels. Stressed lap shear and fracture propagation data will be generated. The detailed test plan for each concept is defined in the appendixes.

4.1.4 Design, Analysis, and Engineering Feasibility Tests

The designs produced during the applications study for all concepts contain areas of engineering unknowns regarding ultimate strength, fatigue resistance, and fracture toughness. These unknowns must be investigated before final design drawings are started. The designs produced in the applications studies will be refined and potential problem areas detailed to provide visibility. Analyses of ultimate strength, fatigue capability, and fracture toughness will be conducted on the refined designs. In those areas where available test data and analysis methods are not sufficient to proceed with final designs, engineering feasibility tests will be conducted. Tests will also be conducted on joint details to determine strength and fatigue requirements, and the optimum geometry of the composite load transfer fittings will be determined. Fracture toughness tests will be conducted on subscale panels for the concept 2 and 3 designs to establish damage containment parameters.

4.1.5 Manufacturing Methods Development

The designs produced during the applications study for all concepts contain fabrication unknowns regarding tolerances and tooling requirements. The development of tolerance control and tooling methods will require that test panels be fabricated and various methods be tried to satisfy the engineering design requirements. These test panels will contain representative components defined by the refined designs produced during phase I. The results of this program will identify the fabrication problem areas and supply preliminary information to

the design engineers. This program will also supply information for fabrication of the engineering feasibility panels, allowables test panels, and the full-scale manufacturing feasibility hardware.

4.1.6 Basic Allowables

Basic engineering allowables required for designing with graphite composites are available in publications such as reference 3. However, due to the variation in mechanical properties caused by fabrication procedures and test methods it is considered necessary to conduct a test program to determine the basic mechanical properties of the selected fiber and resin system. Tests will be conducted to determine the environmental resistance of metal and composite joints and to establish allowable stresses for particular structural configurations in each concept. Tests will also be conducted to determine the effect of flaw size in the cured composite components and to establish quality control limits. The test programs for each concept are defined in the appendixes.

4.1.7 Manufacturing Feasibility Hardware

There are many areas of unknowns regarding the feasibility of fabrication of full-scale components for each design concept. The manufacturing methods development program previously defined will evaluate detail fabrication problems, but the "size effect" of full-scale components will not have been investigated. An example of the size effect is the problems that result during bonding of large components of different relative stiffnesses. The mechanical attachment of stringers that contain residual thermal stresses will also cause some difficulties during fabrication. These problem areas will be investigated by fabricating full-size components and developing the assembly requirements. The results of this study will be incorporated into the final assembly processes for the ground and flight test components. The results of this study will also be used, where necessary, to modify the final designs to ensure a product of acceptable engineering quality. These test panels will contain representative components and, wherever possible, will be tested in the engineering feasibility and verification programs. The feasibility hardware for each concept is defined in the appendixes.

4.1.8 Quality Assurance

Many areas of the concept designs contain structural sections and composite laminate thicknesses that have not previously been subjected to nondestructive inspection. Quality assurance developmental programs will be conducted to establish inspection techniques for these areas.

An optical system for measuring the residual thermal stresses in the concept 1 reinforced stringers will be developed during this program. This system will be assembled, verified, and then released to the fabrication quality assurance group for use in measuring the residual thermal stresses.

The inspection of composite reinforcing on the stringers in all concepts will require the development of through-transmission water-coupled ultrasonic inspection equipment of a size to suit the narrow strips of composite on the stringers. This method of inspection has been selected because of the high flaw resolution that can be obtained. The equipment will be portable and will use small-diameter water nozzles for inspecting the full width of the bonded composite reinforcing strips. The inspection capability of this equipment will be verified by checking reinforced stringer sections with known flaw inclusions.

A section of the thickest keel beam laminate stack in the concept 1 and 2 designs will be inspected to determine the capability of existing through-transmission ultrasonic inspection equipment. Flaws of varying size will be located at various positions through the laminate thickness, and the part will be subjected to ultrasonic inspection. The results of these tests will be used to formulate the inspection procedure for the final parts.

The floor beams will be fabricated as a single-stage bond. The inspection methods required to verify the composite-reinforced chord bond in the area of the web will be investigated. The results of this study will be used to define the inspection procedure for the final parts.

The most suitable techniques for inspecting the flight service aircraft at the end of the 5-year evaluation period will be determined in this phase of the program. Completed panel sections from the manufacturing feasibility studies will be subjected to various inspection methods and selections will be made.

4.1.9 Electrodynamic Systems

The semiconductive nature of the graphite composite components in all design concepts invalidates the electrical bonding, grounding, and shielding methods employed on present all-metal aircraft. The effect of the composite material on lightning protection, static charge dissipation, electrical ground continuity, and electromagnetic interference will have to be evaluated. The results from a recent Air Force contract (ref. 4) will be used in this program. Each concept design will be studied and recommendations made to ensure that all electrical system requirements are satisfied.

4.2 PHASE II—DESIGN, ANALYSIS, AND ENGINEERING VERIFICATION TESTS

4.2.1 Design

The final designs of the ground test and flight service evaluation components will be based on the refined concept designs produced during phase I. The phase I designs will be modified where necessary to incorporate the results of the engineering and manufacturing feasibility tests and the electrical system requirement studies. The titanium surface treatment process and process specifications for composite laminates and adhesives developed during phase I will also be incorporated in the final designs. The design drawings will be produced according to Boeing production drawing standards.

4.2.2 Analysis

The analysis of the components in all concepts will require a detailed check of ultimate strength, fatigue, and fracture toughness. The ultimate strength checks of uniaxial reinforced components will be accomplished by converting the composite to an equivalent metal section and applying conventional analysis methods. The residual thermal stresses will be included in all calculations. A finite-displacement computer model will be established for the complete fuselage section, and all critical load cases will be checked. In the transition areas between the existing fuselage and the composite section, a fine-grid model will be established and used to determine the adequacy of the splice details throughout the operating temperature range of the aircraft. The computer program that will be used for these analyses is described in reference 5. In those areas of cross-plied laminates, a computer program similar to that described in the NAS1-8858 phase III tests will be used (see ref. 6). In the compression-loaded areas, computer analyses will be conducted using the programs defined in references 7, 8, and 9 to check the structural elements for critical buckling loads. A fatigue analysis will be performed, and the resulting fatigue stresses will be compared to the results of the laboratory test panels. A fracture toughness analysis will be conducted to determine damage containment requirements, and the results from the structural test panels will be used for comparison to ensure adequate damage containment capability.

4.2.3 Engineering Verification Tests

Structural tests will be conducted on the most critical components in each design concept to verify the final designs. The results of this program will ensure that the full-scale ground test goals can be achieved. Where the test results indicate a deficiency, the final designs will be modified before the drawings are released for fabrication. The test program for each concept is contained in the appendixes.

4.3 PHASE III—FABRICATION AND QUALITY ASSURANCE

4.3.1 Fabrication

The final fabrication processes will be based on the results of the manufacturing methods development program, the manufacturing feasibility tests, and the final drawings. Basic fabrication processes have been established for all concepts based on studies conducted during the application analysis. These are defined in the appendixes for each concept and are considered valid; only a few detail procedures will be modified by the results of the phase I programs.

4.3.2 Quality Assurance

Final quality assurance procedures for all concepts will be based on the phase I programs and existing inspection methods. Quality assurance procedures have been established for all concepts based on the applications study designs. The following inspection procedures are common to all concepts:

- Standard techniques will be used for all-metal receiving inspection.

- Receiving inspection for the selected composite and the 394° K (250° F) adhesive system will be conducted according to the procurement specification developed in phase I, and receiving inspection of the AF 30 adhesive will be conducted according to existing company specifications.

- Quality control of the composite and 394° K (250° F) adhesive system during fabrication will be maintained according to the process specifications developed in phase I, and AF 30 quality control will be conducted according to existing company specifications.

- Quality control of metal fabrication will be maintained by applying conventional metal inspection techniques.

Particular requirements for each concept are defined in the appendixes.

4.4 PHASE IV—FULL-SCALE GROUND TESTS AND DOCUMENTATION

4.4.1 Full-Scale Ground Tests

Flight evaluation components for all concepts will be subjected to ground tests to verify their structural integrity. These components will be separate from the final flight service evaluation sections, and they will be fabricated from the final drawings. The concept 2 and 3 total fuselage section will be tested to limit and ultimate loads, and a four-lifetime fatigue spectrum will be applied. The concept 1 total fuselage section will not be tested, since only the upper and lower quadrant panels are being modified with the composite components. The major splice areas of the concept 1 panels will be tested in fatigue to verify the final designs. The fatigue resistance and damage containment capability due to pressure loading of the concept 1 panels are not considered unknowns since similar waffle doubler and fail-safe strap construction in the concept 1 designs has already been evaluated in the Boeing 737 aircraft. Blade penetration tests on concept 1 panels will further substantiate the damage containment capabilities. The details of the test programs for each concept are contained in the appendixes.

4.4.2 Documentation

Quarterly reports and a final document for all concepts will be published in accordance with the schedules shown in figures 37, 38, and 39. The quarterly reports will describe the progress that has been made in each area, present test results obtained during the previous 3 months, and define proposed test programs and schedules for the following 3 months. At the completion of the full-scale ground tests, a final document will be prepared that will contain the results of the tests and a summary of the material contained in the quarterly reports. Inspection reports will be submitted at 6-month intervals throughout the flight evaluation period. These reports are described in section 4.5.6.

4.5 PHASE V–FLIGHT SERVICE EVALUATION

4.5.1 Section Installation

All service evaluation components will be assembled in existing production line tools modified to accommodate the small dimensional changes of the composite components. The concept 2 and 3 sections will be assembled outside of the production line and then mated with the adjoining sections in a final assembly. The concept 1 components will be inserted directly into the production line and assembly will proceed without further modification.

4.5.2 Section Instrumentation

Service evaluation sections for all concepts will contain strain gages to monitor the load-carrying capability of the composite-reinforced components. There will be 45 strain gage bridges installed in the quadrant panels of the concept 2 and 3 sections and 30 strain gage bridges installed in he concept 1 panels. The floor beams will be instrumented with an additional five strain gage bridges in each concept. The wire leads will be routed to one location where a flight recorder can be conveniently connected.

4.5.3 Aircraft Certification

The service evaluation aircraft will be subjected to the standard flight test program for all production line aircraft. No additional flight tests will be required for FAA certification, since the weight decrease and altered stiffness are not sufficient to affect the flight characteristics. During the normal flight test for airline acceptance, the aircraft will be subjected to two maneuver conditions to obtain strain gage data. The first maneuver will be performed to provide tension in the upper quadrant panel and compression in the lower quadrant panel. The second maneuver will be performed to provide compression in the upper quadrant panel and tension in the lower quadrant panel. During these maneuvers, other data related to load factor, airspeed, altitude, and gross weight distribution will be recorded. This information will become baseline data that will be used for comparison throughout the flight evaluation period.

4.5.4 Aircraft Monitoring

At 6-month intervals for a period of 5 years, the service evaluation aircraft will be subjected to the two maneuvers previously defined. Strain gage data will be obtained during these tests and the aircraft will then be returned to commercial service.

The strain gage data obtained during each of the 6-month checks will be compared to the baseline data. If the strain gage data indicate that the composite reinforcing is correctly loaded, the aircraft will be allowed to continue in commercial service. If, however, these data indicate significant change in the load levels, the aircraft will be recalled and a detailed inspection performed.

4.5.5 Aircraft Inspection and Disposition

At the end of the 5-year service evaluation program, the composite components will be nondestructive inspected for voids, delamination, and environmental degradation. This inspection will require that the interior panels and insulation liners be removed. At the completion of this inspection, and depending upon the results of the total evaluation program, the composite components will either remain on the aircraft or be replaced with conventional structure and the aircraft will be returned to commercial service.

4.5.6 Inspection Reports

Inspection reports will be prepared at 6-month intervals throughout the flight evaluation period. These reports will contain the strain gage information obtained on the previous flight test and the utilization data of the aircraft for the previous 6 months. At the end of the flight service evaluation period a report will be submitted that contains the results of the final inspection and a summary of all data collected during the 6-month inspections.

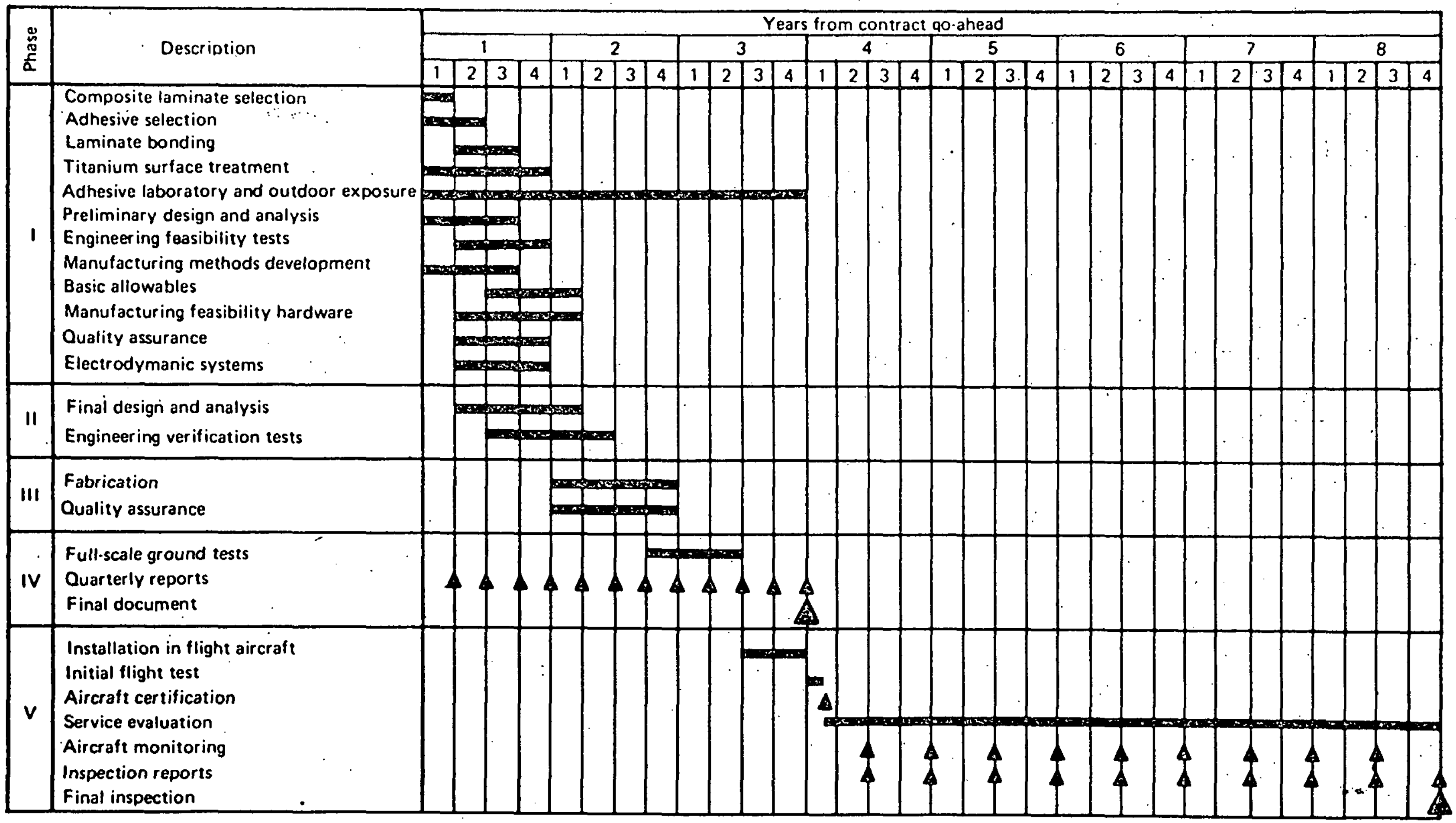


FIGURE 37.—CONCEPT 1—DEVELOPMENTAL PROGRAM SCHEDULE

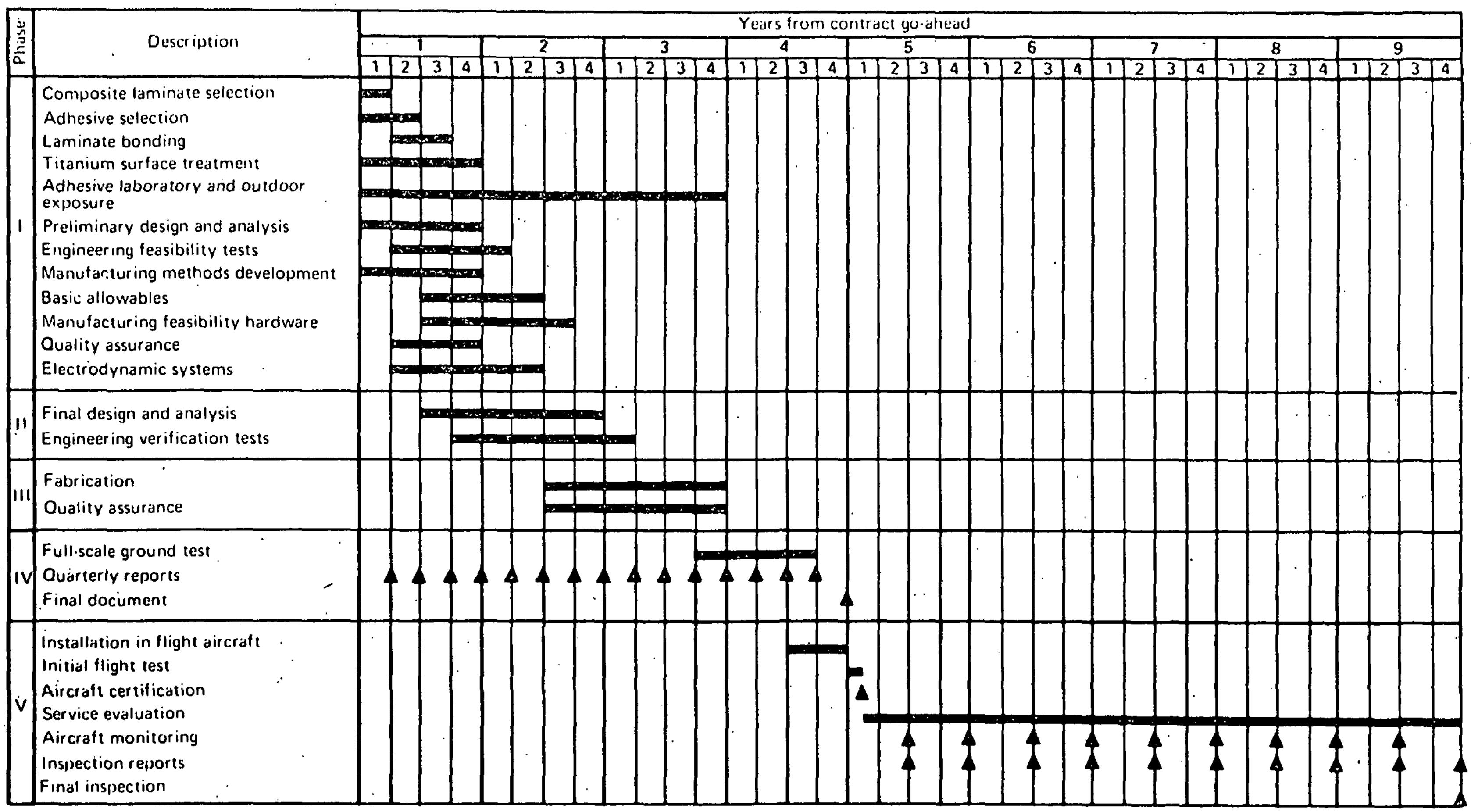


FIGURE 38.—CONCEPT 2—DEVELOPMENTAL PROGRAM SCHEDULE

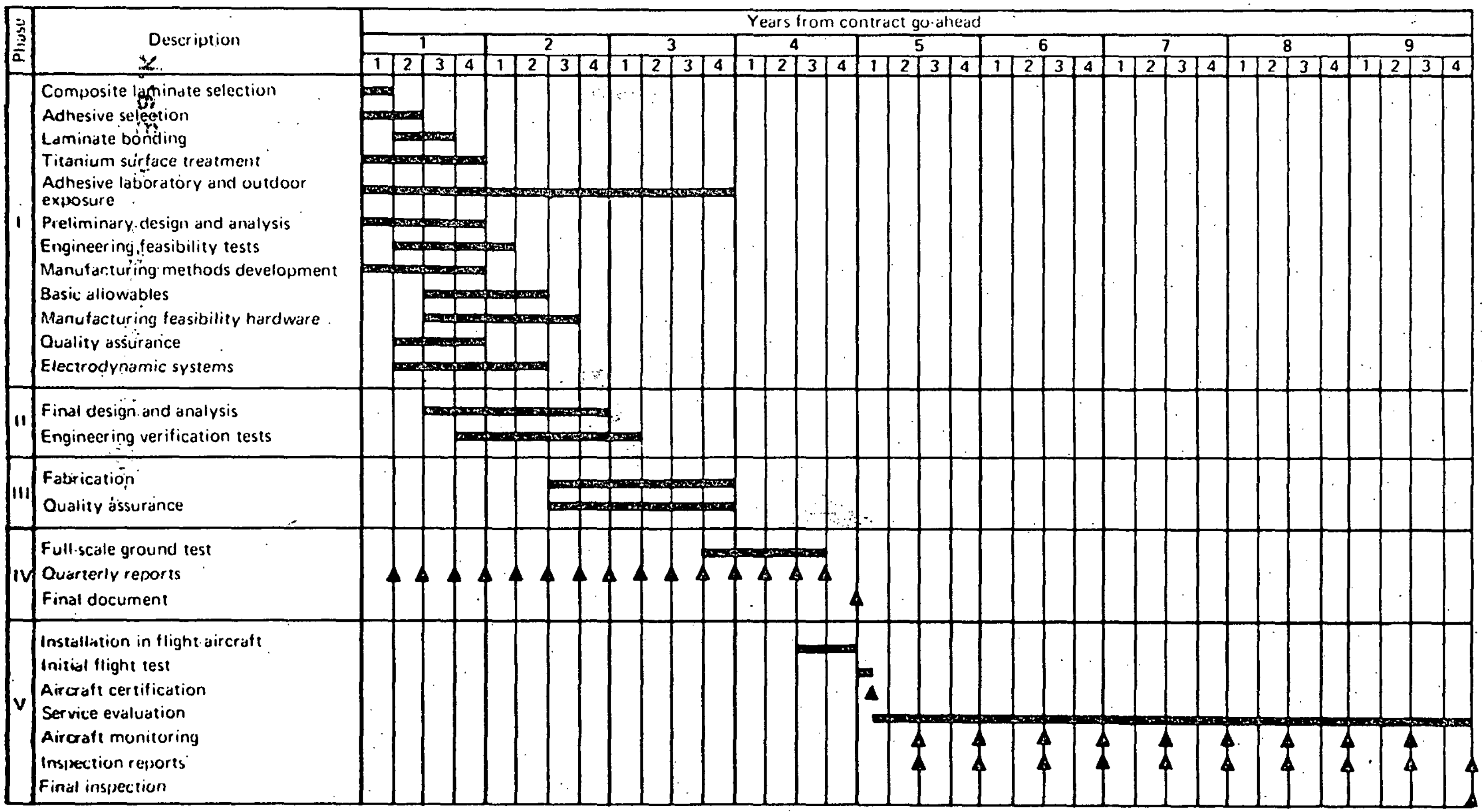


FIGURE 39.—CONCEPT 3—DEVELOPMENTAL PROGRAM SCHEDULE

APPENDIX A
DETAILED TEST AND MANUFACTURING PLANS FOR THE CONCEPT 1 DEVELOPMENT PROGRAM

A.1 LAMINATE SELECTION TESTS

The composite laminates that were selected during the materials study (sec. 3.3) will be evaluated in the following test program. One composite system will be selected for use throughout the developmental program based on the results of this test program.

Tension, compression, and interlaminar shear tests will be conducted on uniaxial composite at three temperatures, and interlaminar shear tests of bonded titanium and composite will be conducted. This test program is summarized in table A-1.

A.2 ADHESIVE LABORATORY AND OUTDOOR EXPOSURE TESTS

Laboratory and outdoor exposure tests will be conducted using AF 30 and the selected 394° K (250° F) system. Stressed lap shear and fracture propagation data will be generated. The fracture propagation specimen is shown in figure A-1, the lap shear specimen in figure A-2, and the detailed test plan in table A-2.

A.3 BASIC ALLOWABLES

The basic allowables test program is shown in table A-3.

A.4 MANUFACTURING FEASIBILITY HARDWARE

Full-scale components will be fabricated during this program. These components are defined as follows:

- A 3050-mm (120-in.) long by 610-mm (24-in.) wide section of the upper quadrant skin will be fabricated. A 3050-mm (120-in.) long stringer of the stiffest configuration will be fabricated and riveted to the skin.
- A 3050-mm (120-in.) long by 610-mm (24-in.) wide section of the lower quadrant skin will be fabricated. A 3050-mm (120-in.) long stringer of the stiffest configuration will be fabricated and riveted to the skin.
- A 2540-mm (100-in.) long section of a keel beam chord will be fabricated and fastened to the lower quadrant skin panel.

A.5 ENGINEERING VERIFICATION TESTS

The following programs define the engineering verification tests that will be performed for the concept 1 development program.

A.5.1 Test 1—Residual Thermal Stress Fatigue Test

This test will evaluate the effect on the fatigue life of residual thermal stresses in reinforced stringers. The section of the panel is shown in figure A-3; the panel is 910 mm (36 in.) long. The number of cycles to failure will be obtained for 12 specimens—six in each of two stringer configurations (5 and 20 plies, 0°). Three specimens of each configuration will be tested at 218° K (-67° F) and three at room temperature.

A.5.2 Test 2—Lower Quadrant Compression Panel

This test will evaluate the compression load capability of a representative section of the lower quadrant design (fig. A-4). The panel will be tested by loading the keel beams with concentrated loads and reacting these with a uniform load along the other panel edge. The panel will be curved, and the unloaded edges will be simply supported. Two panels will be tested to compression ultimate. The instrumentation on each panel will consist of 40 axial gages and 10 rosette gages.

A.5.3 Test 3—Axial Tension Fail-Safe Panel

This test will evaluate the fatigue crack propagation rates and damage containment capability of a representative upper quadrant panel under axial tension load (fig. A-5). Fatigue cracks will be cut in the panel, the panel will be fatigue cycled for approximately 50 000 cycles, and crack growth rates will be measured. The fatigue cracks will be repaired and blade penetration tests conducted on the panel under axial tension. A total of five panels will be tested with two fatigue crack propagation tests and two blade tests on each panel. The instrumentation on each panel will consist of 40 axial gages and 20 rosette gages.

A.6 FABRICATION PROCESSES

The fabrication processes for the upper and lower quadrant panels are defined as follows:

- The composite will be cured and bonded to the titanium load transfer fitting with AF 30 adhesive as a subassembly.
- The composite and load transfer fitting will be bonded to the aluminum stringer with the 394° K (250° F) adhesive. The stringer will be restrained by a steel tool during bonding and lateral support will be provided to prevent buckling.

- The waffle doubler will be bonded to the skin and the waffle pattern will be formed by chemical milling.

- Reinforcing doublers will be envelope bagged and bonded to the skin assembly with the 394° K (250° F) adhesive.

- The stringers will be riveted to the skin assemblies by conventional processes.

The fabrication processes for the keel beam chords are defined as follows:

- The composite plies and titanium end fittings will be bonded with AF 30 adhesive as a subassembly. Bonding will be done on surfaces that simulate the metal parts.

- The composite and end fittings will be bonded to the aluminum section with the 394° K (250° F) adhesive.

The fabrication of the floor beams is defined as follows:

- The composite chords will be cured as a subassembly.

- The composite, cap strips, web channel sections, doublers, and core will be bonded with the 394° K (250° F) adhesive.

A.7 QUALITY ASSURANCE FOR FABRICATION

During each phase of fabrication, nondestructive inspection will be performed on all components. The fabrication sequences have been arranged so as to provide easy access to all bond lines. Water-coupled through-transmission ultrasonic inspection will be the primary process used for quality assurance control because of its high resolution of flaw detection. The detailed quality control procedures for the concept 1 components are defined as follows:

- The cured stringer reinforcing laminate and the step fitting bond will be inspected as a subassembly.

- The composite laminate and end fitting bond to the stringer will be inspected as a subassembly.

- The verification of the required level of residual thermal stress and its uniformity will be accomplished by using the optical comparator developed in phase I.

- The keel beam composite laminate and load transfer fittings will be inspected as a subassembly and the bond between the laminate and aluminum sections will be inspected after final assembly.

- Quality assurance methods for the floor beam assembly will be defined by the developmental program in phase I.

- Quality assurance of the fastener installation and final assembly will be controlled by standard inspection procedures.

A.8 FULL-SCALE COMPONENT TESTS

The full-scale component tests that will be conducted for the concept 1 design are defined as follows.

A.8.1 Test 1—Body Station 740 and 480 Crown Splice Fatigue Tests

These tests will evaluate the fatigue capability of the BS 740 and 480 stringer and skin splice details (fig. A-6). Three test panels that represent each area will be fatigue tested to failure. The instrumentation on each panel will consist of 20 axial gages.

A.8.2 Test 2—Body Station 680 Lower Quadrant Splice Compression Test

This test will evaluate the ultimate compression capability of the BS 680 compression splice (fig. A-7). Three test panels that represent this splice area will be tested to failure. The instrumentation on each panel will consist of 40 axial gages.

A.8.3 Test 3—Pressure-Loaded Fail-Safe Panel

This test will evaluate the damage containment capability of a representative fuselage panel under pressure load (fig. A-8). The test panel will be mounted in a section of conventional fuselage structure and loaded by internal pressure. A total of three panels will be tested with four blade penetration shots on each panel. The instrumentation on each panel will consist of 40 axial gages and 20 rosette gages.

TABLE A-1.–LAMINATE SELECTION TEST PROGRAM–CONCEPT 1

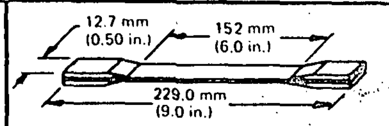
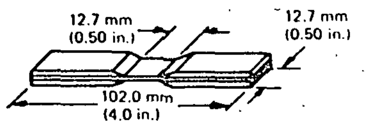
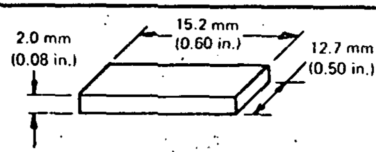
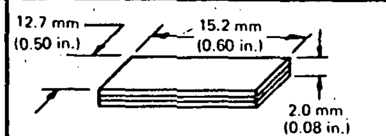
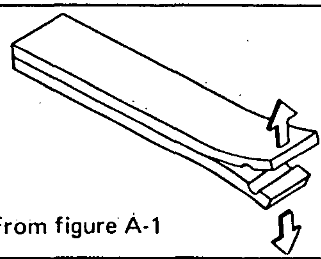
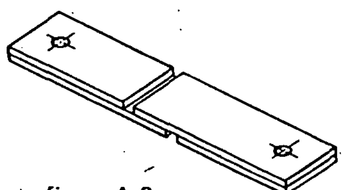
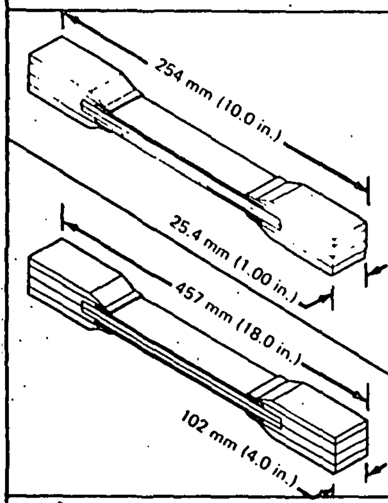
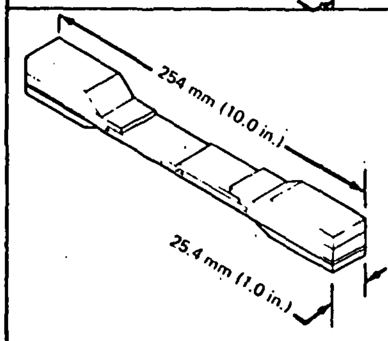
Type of test	Specimen configuration	Test objective	Fabrication details	Test temperature	Composites	Replicates	Total
Tension		Modulus ultimate strength	Unidirectional 0° six plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Compression		Modulus ultimate strength	Unidirectional 0° six plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Interlaminar shear		Shear stress	Unidirectional 0° 16 plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Interlaminar shear composite metal bond		Bond shear	Composite six plies Titanium 0.50 mm (0.020 in.)	RT, 218°K (-67°F), 344°K (160°F)	3	6	54

TABLE A-2.—ADHESIVE LABORATORY AND OUTDOOR EXPOSURE TEST PROGRAM—CONCEPT 1

Type of test	Specimen configuration	Adhesives	Adherend sets	Specimens		Stress levels	Total specimens exposed	Control specimens
				Outdoor exposure	Laboratory exposure			
Fracture propagation	 From figure A-1	2	Al-Al Al-comp Ti-comp	4	4	1	48	20
Lap shear	 From figure A-2	2	Al-Al Al-comp Ti-comp	4	4	[a]4	192	50

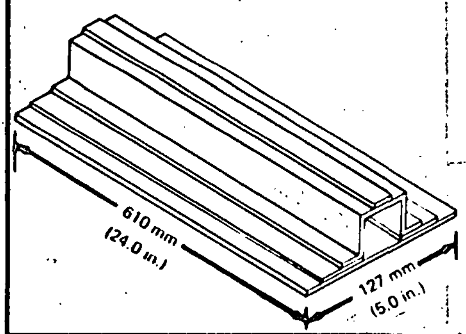
[a]Stress levels 0%, 10% 20%, and 30% of the 344°K (160°F) allowable

TABLE A-3.–BASIC ALLOWABLES TEST PROGRAM–CONCEPT 2

Specimen configuration	Type of test	Test objective	Fabrication details	Variables	Replicates	Total
	Tension ultimate	Obtain ultimate tension load at failure	Unidirectional 0° plies *Bonded titanium fitting* Three laminates (4, 8, 16 plies)	Three thicknesses Three temperatures 218°K (-67°F) RT 344°K (160°F)	3	27
	Fatigue	Obtain number of cycles to failure	Unidirectional 0° plies Bonded titanium fitting Two laminates (4, 16 plies)	Two thicknesses Two temperatures 218°K (-67°F) RT Two stress levels	3	24
	Environmental cycling	Obtain number of environmental exposure cycles to failure[a]	*Unidirectional* 0° plies Bonded titanium fitting Two laminates (4, 16 plies)	Two thicknesses	10	20
	Fatigue with voids	Obtain number of cycles to failure	Unidirectional 0° plies Bonded titanium fitting with void inclusions Two laminates (4, 8 plies)	Two thicknesses Four void sizes	3	24
	Fatigue of tapered laminates	Obtain number of cycles to failure	Unidirectional 0° plies Bonded titanium fitting on aluminum Ends unsymmetric Two aluminum gages 0.63 mm (0.025 in.) 1.02 mm (0.04 in.) Two ply configurations	Four thicknesses Two stress levels	3	24
	Environmental cycling	Obtain number of environmental exposure cycles to failure[a]	*Unidirectional* 0° plies Bonded titanium fitting on aluminum Ends unsymmetric One aluminum gage 0.63 mm (0.025 in.) Two ply configurations	Two thicknesses	5	10

[a]Environmental cycle: 218°K (-67°F) to 344°K (160°F) at 100% relative humidity; stressed at 30% of the 344°K allowable.

TABLE A-3.—Concluded

Specimen configuration	Type of test	Test objective	Fabricated details	Variables	Replicates	Total
	Compression ultimate	Obtain ultimate compression load at failure	Unidirectional 0° plies Bonded waffle doubler Mechanically fastened aluminum stringer Two stringer gages 0.91 mm (0.036 in.) 1.02 mm (0.040 in.) Two laminates (5, 20 plies)	Four thicknesses	3	12
	Compression ultimate with voids	Obtain ultimate compression load at failure with void inclusions	Unidirectional 0° plies Bonded waffle doubler Mechanically fastened aluminum stringer Stringer gage 1.02 mm (0.040 in.) Two laminates (5, 20 plies) Three void sizes	Two thicknesses Three void sizes	3	18

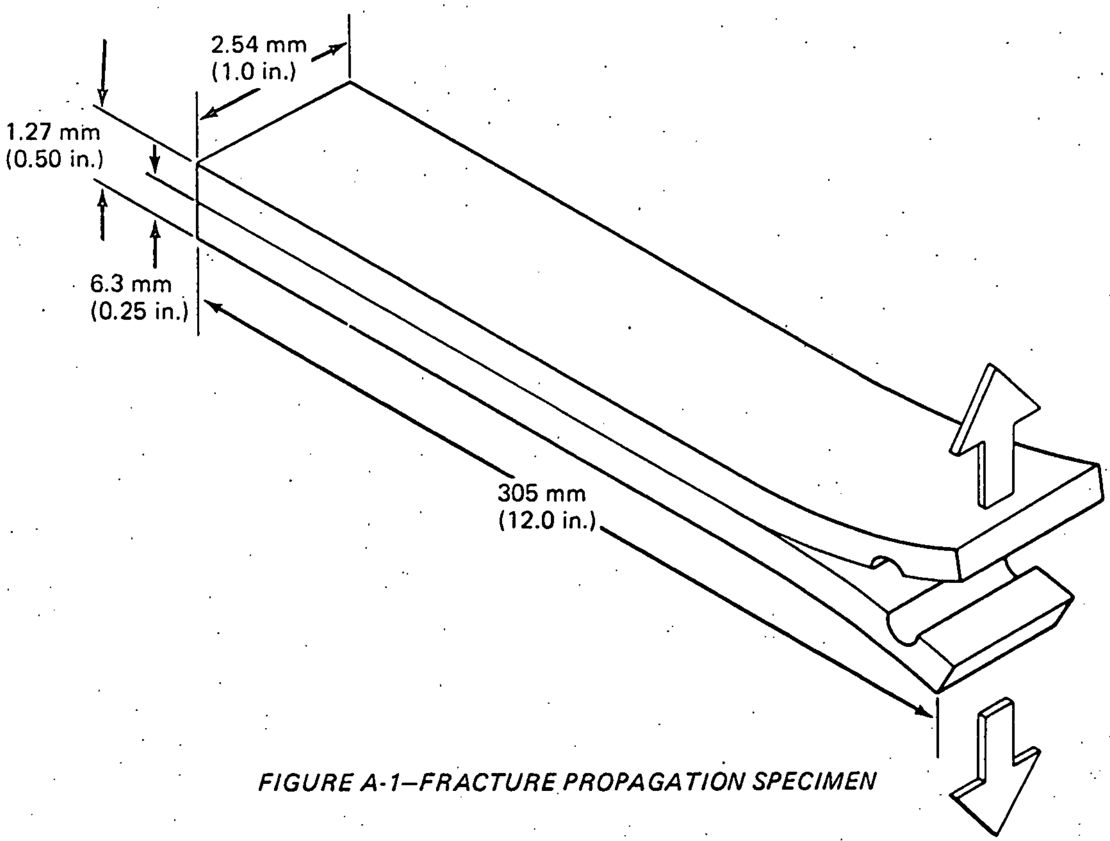


FIGURE A-1—FRACTURE PROPAGATION SPECIMEN

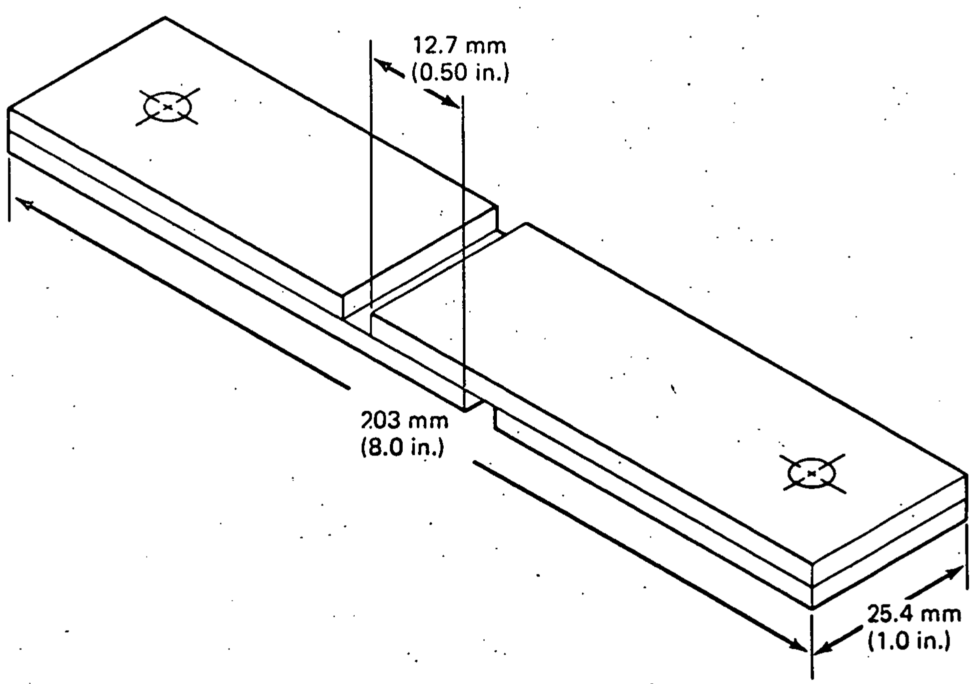


FIGURE A-2—LAP SHEAR SPECIMEN

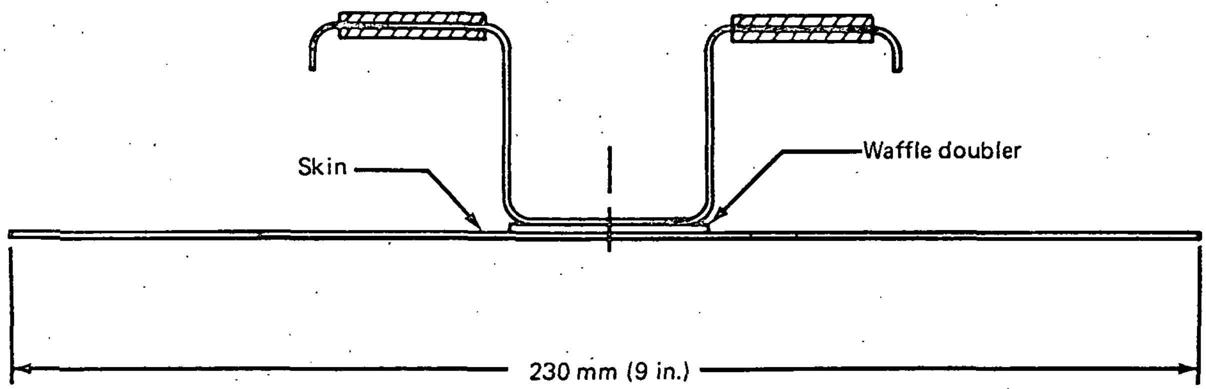


FIGURE A-3—RESIDUAL THERMAL STRESS FATIGUE TEST SPECIMEN SECTION—CONCEPT 1

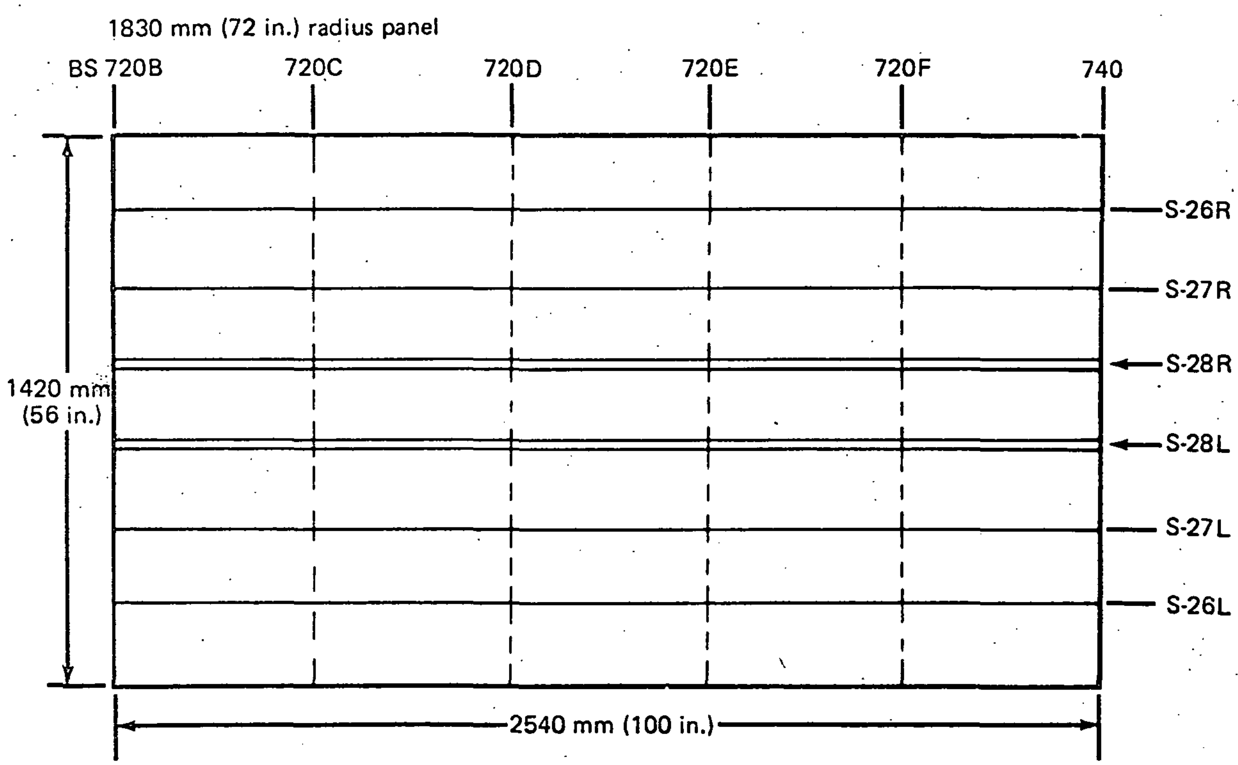


FIGURE A-4.—LOWER QUADRANT COMPRESSION PANEL TEST SPECIMEN CONFIGURATION—CONCEPT 1

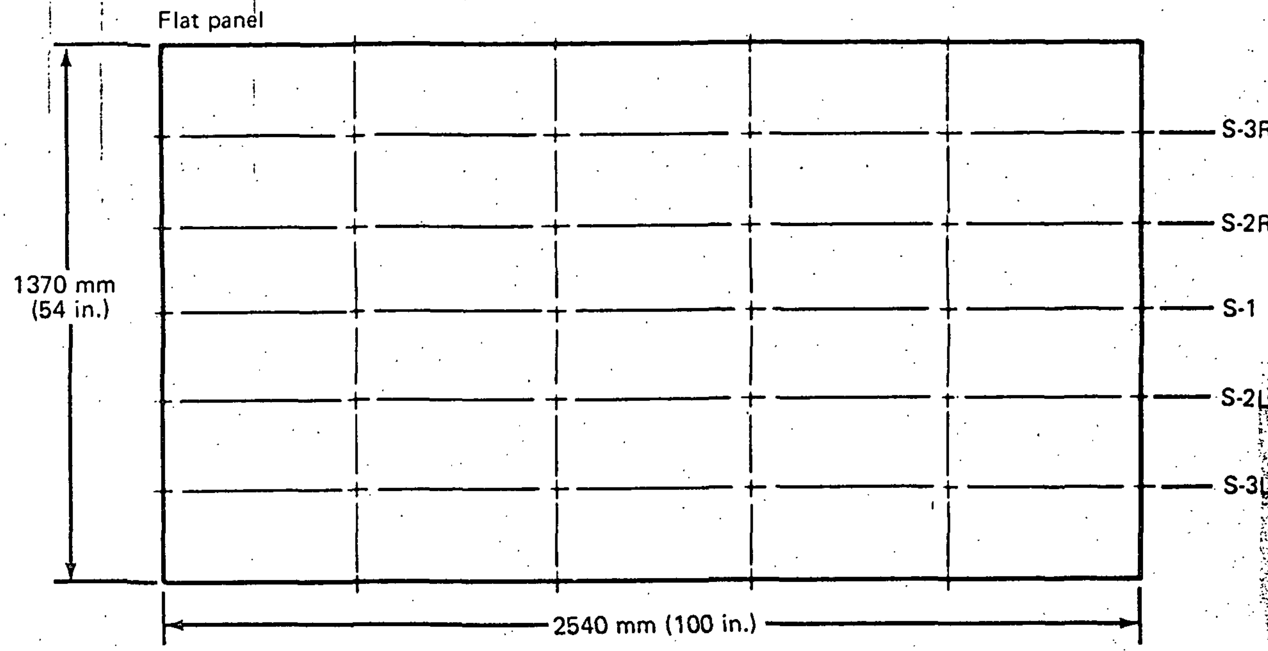


FIGURE A-5.—AXIAL TENSION FAIL-SAFE PANEL TEST SPECIMEN CONFIGURATION—CONCEPT 1

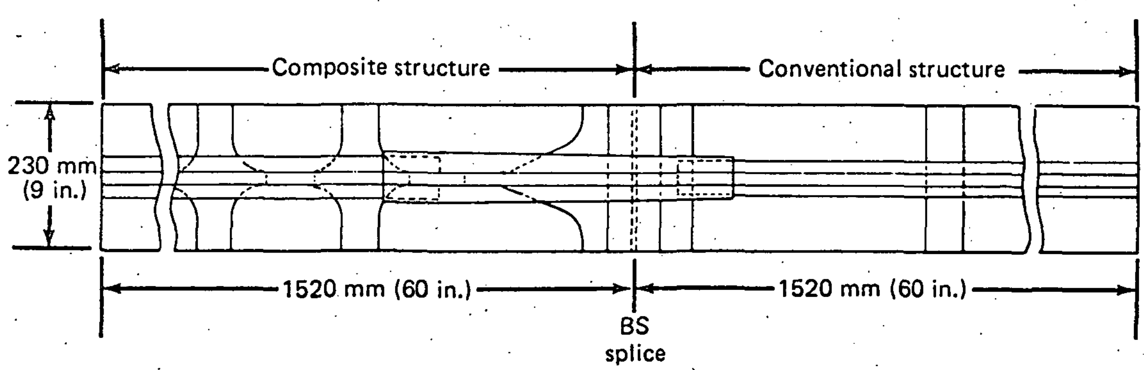


FIGURE A-6.—BODY STATIONS 740 AND 480 CROWN SPLICE FATIGUE TEST SPECIMEN CONFIGURATION—CONCEPT 1

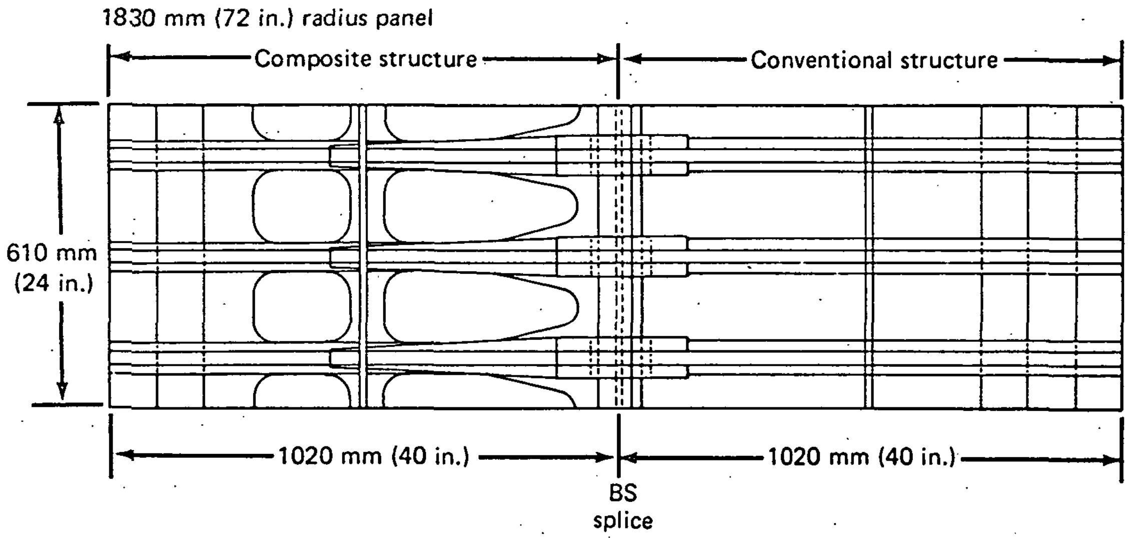


FIGURE A-7.—BODY STATION 680 LOWER QUADRANT SPLICE COMPRESSION TEST SPECIMEN CONFIGURATION—CONCEPT 1

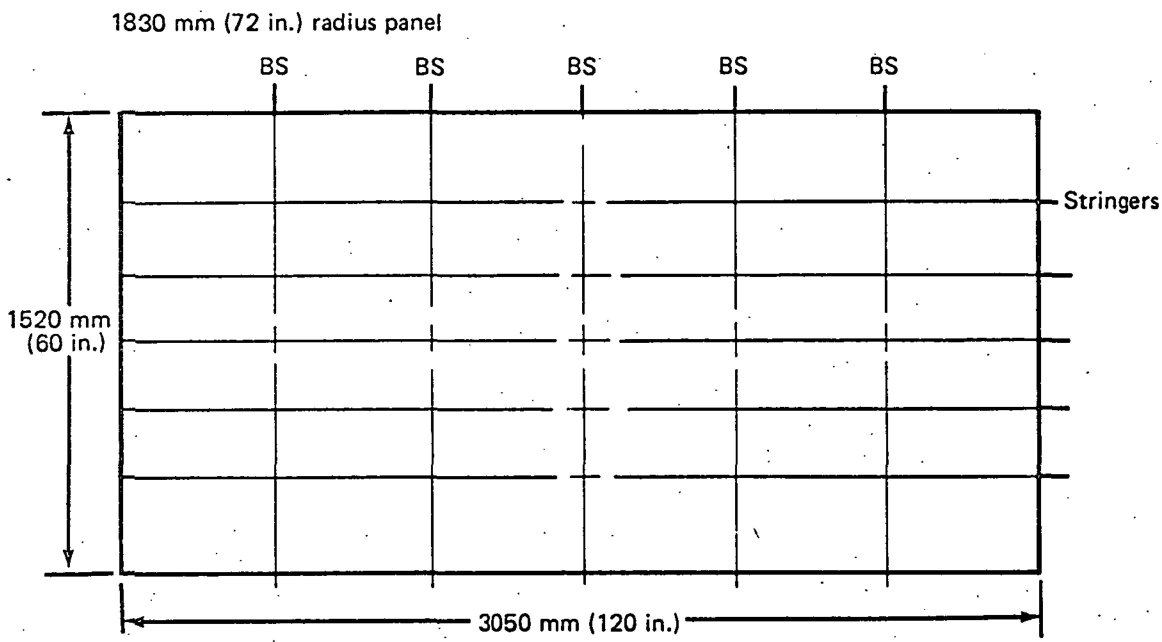


FIGURE A-8.—PRESSURE-LOADED FAIL-SAFE PANEL TEST SPECIMEN CONFIGURATION

APPENDIX B
DETAILED TEST AND MANUFACTURING PLANS FOR THE CONCEPT 2 DEVELOPMENT PROGRAM

B.1 LAMINATE SELECTION TESTS

The composite laminates that were selected during the materials study (sec. 3.3) will be evaluated in the following test program. One high-strength composite system will be selected for use throughout the developmental program based on the results of this test program.

Tension, compression, and interlaminar shear tests will be conducted on uniaxial composite at three temperatures, and interlaminar shear tests of bonded titanium and composite will be conducted. In-plane shear tests of ±45° plies will also be conducted. This test program is summarized in table B-1.

B.2 ADHESIVE LABORATORY AND OUTDOOR EXPOSURE TESTS

Laboratory and outdoor exposure tests will be conducted using AF 30 and the selected 394° K (250° F) system. Stressed lap shear and fracture propagation data will be generated. The fracture propagation specimen is shown in figure B-1, the lap shear specimen in figure B-2, and the detailed test plan in table B-2.

B.3 BASIC ALLOWABLES

The basic allowables test program is shown in table B-3.

B.4 MANUFACTURING FEASIBILITY HARDWARE

Full-scale components will be fabricated during this program. These components are defined as follows:

- A 3050-mm (120-in.) long by 910-mm (36-in.) wide section of the upper quadrant containing the titanium skin, the composite skin reinforcing, three reinforced stringers, the frame tees, and frame J-sections.
- A 3050-mm (120-in.) long by 910-mm (36-in.) wide section of the window belt containing the titanium skin, the composite skin reinforcing, the window frames, STR 11 and 13, the frame tees, and frame J-sections.
- A 3050-mm (120-in.) long by 910-mm (36-in.) wide section of the side quadrant below the window belt containing the titanium skin, the composite skin reinforcing, the STR 17 mechanical splice, the frame tees, and frame J-sections.
- A 3050-mm (120-in.) long by 690-mm (27-in.) wide section of the lower quadrant center skin containing the titanium skin, one keel beam chord, the frame tees, and frame J-sections.

B.5 ENGINEERING VERIFICATION TESTS

The following programs define the engineering verification tests that will be performed for the concept 2 development program.

B.5.1 Test 1—Residual Thermal Stress Fatigue Test

This test will evaluate the effect on the fatigue life of residual thermal stresses in reinforced stringers. The section of the panel is shown in figure B-3; the panel is 910 mm (36 in.) long. The number of cycles to failure will be obtained for 12 specimens—six in each of two stringer configurations (5 and 20 plies, 0°). Three specimens of each configuration will be tested at 218° K (-67° F) and three at room temperature.

B.5.2 Test 2—Bonded Shear Web Stiffeners

This test will evaluate the bond and stiffness requirements of frame tees and stringers for stabilizing the body skin shear panels (fig. B-4). The panel is a representative section of the skin above or below the window panel that contains frame tees, stringers, and composite-reinforced titanium skin.

Panels will be fatigue tested by applying a cyclic load to a cantilever beam and recording the cycles to failure and mode of failure for eight specimens—four in each of two shear web gages. Two stiffener bond lap lengths will be tested. The instrumentation on each panel will consist of eight axial gages and 15 rosette gages.

B.5.3 Test 3—Window Panel Ultimate and Fatigue Tests

This program will evaluate the shear and pressure load capability of a window belt panel (fig. B-5). The window panel will be tested with shear and pressure loads. An ultimate test will be performed on one panel, and simulated fatigue cycles will be applied to a second panel. The test panel is a representative section of the window belt area. Panels will be loaded to ultimate pressure and shear loads so as to obtain strain surveys, ultimate loads, modes of failure, and number of cycles to failure. The instrumentation on each panel will consist of 15 axial gages, 10 crack-wire circuits, 40 rosette gages, and five deflection indicators.

B.5.4 Test 4—Lower Quadrant Compression Panel

This test will evaluate the compression load capability of a representative section of the lower quadrant design (fig. B-6). The panel will be tested by loading the keel beams with concentrated loads and reacting these with a uniform load along the other panel edge. The panel will be curved and the unloaded edges will be simply supported. Two panels will be tested to compression ultimate. The instrumentation on each panel will consist of 40 axial gages and 10 rosette gages.

B.5.5 Test 5—Axial Tension Fail-Safe Panel

This test will evaluate the fatigue crack propagation rates and damage containment capability of a representative upper quadrant panel under axial tension load (fig. B-7). Fatigue cracks will be cut in the panel, the panel will be fatigue cycled for approximately 50 000 cycles, and crack growth rates will be measured. The fatigue cracks will be repaired and blade penetration tests conducted on the panel under axial tension. A total of five panels will be tested with two fatigue crack propagation tests and two blade tests on each panel. The instrumentation on each panel will consist of 40 axial gages and 20 rosette gages.

B.5.6 Test 6—Frame Tee Fail-Safe Strap Panel Test

This test will evaluate the fatigue crack propagation rates and damage containment capability of a representative quadrant panel under pressure load (fig. B-8). Fatigue cracks will be cut in the panel, the panel will be fatigue cycled for approximately 50 000 cycles, and crack growth rates will be measured. The fatigue cracks will be repaired and blade penetration tests conducted on the panel under hoop pressure tension loads. A total of five panels will be tested with two fatigue crack propagation tests and two blade tests on each panel. The instrumentation on each panel will consist of 40 axial gages and 20 rosette gages.

B.5.7 Test 7—Damage Containment with Biaxial Stresses

This test will evaluate the damage containment capability of a fuselage skin panel under pressure stress and axial tension or compression (fig. B-9). A representative section of a skin panel will be installed in a section of a 727 fuselage contained between two pressure bulkheads. The fuselage section will be pressurized and axial tension or compression will be introduced into the shell by hydraulic jacks acting between the end bulkheads. Blade penetration tests will be conducted on the test panel under biaxial stress. Five panels will be tested with four blade shots in each panel. The instrumentation on each panel will consist of 40 axial gages and 20 rosette gages.

B.5.8 Test 8—Body Station 740 and 680 Crown Splice Fatigue Tests

These tests will evaluate the fatigue capability of the BS 740 and 680 stringer and skin splice details (fig. B-10). Three test panels that represent each area will be fatigue tested to failure. The instrumentation on each panel will consist of 20 axial gages.

B.5.9 Test 9—Body Station 680 Lower Quadrant Splice Compression Test

This test will evaluate the ultimate compression capability of the BS 680 compression splice (fig. B-11). Three test panels that represent this splice area will be tested to failure. The instrumentation on each panel will consist of 40 axial gages.

B.6 FABRICATION PROCESSES

The fabrication processes for the upper quadrant panel are defined as follows:

- The composite will be cured and bonded to the titanium load transfer fitting with AF 30 adhesive as a subassembly.

- The composite skin reinforcing will be cured and bonded to the precurved picture frame load transfer fitting with AF 30 adhesive on a flat tool surface. The assembly will then be vacuum bagged and heated to 394° K (250° F) in a curved bond tool.

- The composite skin reinforcing will be bonded to the titanium skin in a curved bond tool with the 394° K (250° F) adhesive.

- The stringers, stringer reinforcing, frame tees, and skin assembly will be bonded with the 394° K (250° F) adhesive.

The fabrication processes for the window belt panel are defined as follows:

- The skin reinforcing will be bonded in the recesses with the 394° K (250° F) adhesive in a curved bond tool.

- The overall skin reinforcing will be bonded to the precurved picture frame with AF 30 adhesive on a flat tool. The assembly will then be vacuum bagged and heated to 394° K (250° F) in a curved bond tool.

- The composite skin will be bonded to the titanium skin assembly with the 394° K (250° F) adhesive in a curved bond tool.

- The window frames, frame tees, and stringers will be bonded to the skin panel assembly with the 394° K (250° F) adhesive.

The fabrication processes for the lower quadrant center skin panel are defined as follows:

- The tapered doubler and the frame tees will be bonded to the skin with the 394° K (250° F) adhesive.

- The keel beam filler doubler will be bonded to the skin assembly with the 394° K (250° F) adhesive.

- The keel beam chords will be mechanically attached to the panel assembly.

The fabrication processes for the keel beam chords are defined as follows:

- The composite plies and titanium end fittings will be bonded with AF 30 adhesive as a subassembly. Bonding will be done on surfaces that simulate the metal parts.

- The composite and end fittings will be bonded to the titanium sections with the 394° K (250° F) adhesive.

The fabrication processes for the floor beam are defined as follows:

- The composite chords will be cured as a subassembly.

- The composite, cap strips, web channel sections, doublers, and core will be bonded with the 394° K (250° F) adhesive.

B.7 QUALITY ASSURANCE FOR FABRICATION

During each phase of fabrication, nondestructive inspections will be performed on all components. The fabrication sequences have been arranged so as to provide easy access to all bond lines. Water-coupled through-transmission ultrasonic inspection will be the primary process used for quality assurance control because of its high resolution of flaw detection. The detailed quality control procedures for the concept 2 components are defined as follows:

- The cured stringer and skin-reinforcing laminate and the step fitting bonds will each be inspected as subassemblies.

- The cured and bonded laminate in the recesses in the side quadrant will be inspected as a subassembly.

- The bond of the composite skin and step fitting to the titanium skin will be inspected as a subassembly.

- The frame tee, stringer, and stringer-reinforcing bonds to the skin panels will be inspected between each bonding sequence.

- The keel beam composite laminate will be inspected as a subassembly, and the bond between the laminate and titanium sections will be inspected after final assembly.

- Quality assurance methods for the floor beam assembly will be defined by the developmental program in phase I.

- Quality assurance of the fastener installation and final assembly will be controlled by standard inspection procedures.

B.8 FULL-SCALE GROUND TEST

The full-scale ground test will consist of a complete concept 2 fuselage section 4570-mm (180-in.) long with a 2030-mm (80-in.) long transition section of conventional 727 structure on each end. The test fuselage will be mounted as a cantilever beam, and the loads will be introduced through a loading boom. The test setup is shown in figure B-12. Compressed air will be used for fuselage pressurization and styrofoam will be used to reduce the air volume. The static loads will consist of seven conditions of positive and negative bending moment, ground handling, and pressure. The fatigue loading will be a ground-air-ground

spectrum consisting of taxi, positive vertical, negative vertical, and pressure loading. Magnetic digital tape programs will be used to control the loads. A computer-controlled digital data acquisition system will be used to record all data. The following instrumentation will be used on this test:

- 150 axial strain gages
- 60 rosette strain gages
- 15 electrical deflection indicators
- 10 crack-wire circuits
- 1.86 sq m (20 sq ft) of photoelastic coating located in 10 separate locations
- 5 inertia-triggered strobe flash camera circuits
- 10 sound detection circuits

During cyclic fatigue testing, four major inspections will be performed for each lifetime. The test program sequence is as follows.

- Load to 75% limit load, obtain strain survey
- Conduct cyclic fatigue test, four lifetimes
- Conduct seven static load surveys to limit load
- Conduct seven static load conditions to ultimate

TABLE B-1.—LAMINATE SELECTION TEST PROGRAM—CONCEPT 2

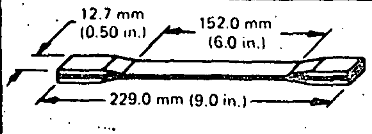
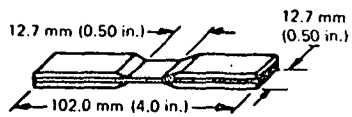
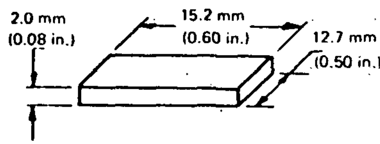
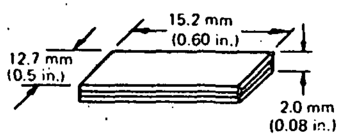
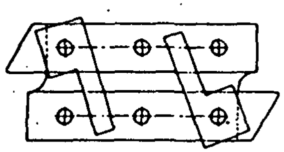
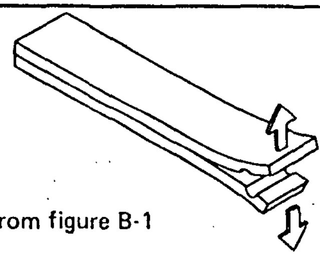
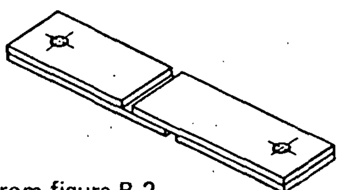
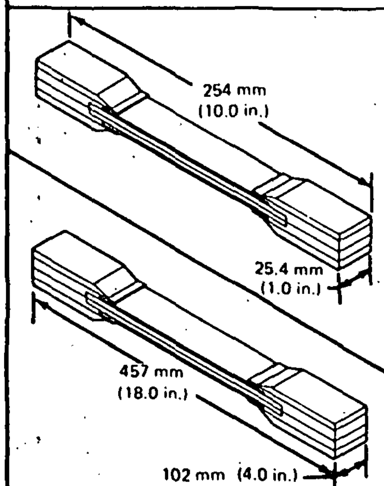
Type of test	Specimen configuration	Test objective	Fabrication details	Test temperature	Composites	Replicates	Total
Tension		Modulus, ultimate strength	Unidirectional 0° six plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Compression		Modulus ultimate strength	Unidirectional 0° six plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Interlaminar shear		Shear stress	Unidirectional 0° 16 plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Interlaminar shear composite metal bond		Bond shear	Composite six plies Titanium 0.50 mm (0.020 in.)	RT, 213°K (-67°F), 344°K (160°F)	3	6	54
In-plane shear		In-plane shear strength	± 45° laminates four plies	RT	3	6	18

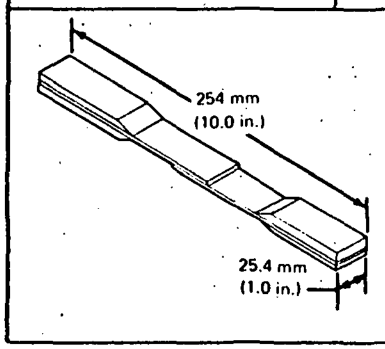
TABLE B-2.–ADHESIVE LABORATORY AND OUTDOOR EXPOSURE TEST PROGRAM–CONCEPT 2

Type of test	Specimen configuration	Adhesives	Adherend sets	Specimens		Stress levels	Total specimens exposed	Control specimens
				Outdoor exposure	Laboratory exposure			
Fracture propagation	 From figure B-1	2	Ti-Ti Comp-comp Ti-comp	4	4	1	48	20
Lap shear	 From figure B-2	2	Ti-Ti Comp-comp Ti-comp	4	4	[a]4	192	50

[a]Stress levels 0%, 10%, 20%, and 30% of the 344°K (160°F) allowable

TABLE B-3.—BASIC ALLOWABLES TEST PROGRAM—CONCEPT 2





Specimen configuration	Type of test	Test objective	Fabrication details	Variables	Replicates	Total
	Tension ultimate	Obtain ultimate tension load at failure	Unidirectional 0° plies Bonded titanium fitting Three laminates (4, 8, 16 plies)	Three thicknesses Three temperatures 218°K (-67°F) RT 344°K (160°F)	3	27
	Fatigue	Obtain number of cycles to failure	Unidirectional 0° plies Bonded titanium fitting Two laminates (4, 16 plies)	Two thicknesses Two temperatures 218°K (-67°F) RT Two stress levels	3	24
	Environmental cycling	Obtain number of environmental exposure cycles to failure[a]	Unidirectional 0° plies Bonded titanium fitting Two laminates (4, 16 plies)	Two thicknesses	10	20
	Fatigue with voids	Obtain number of cycles to failure	Unidirectional 0° plies Bonded titanium fitting with void inclusions Two laminates (4, 8 plies)	Two thicknesses Four void sizes	3	24
	Tension ultimate	Obtain ultimate tension load at failure	Bonded titanium lap joint Three gages 0.38 mm (0.015 in.) 0.51 mm (0.020 in.) 1.02 mm (0.040 in.)	Three lap lengths Three temperatures 218°K (-67°F) RT 344°K (160°F)	3	27
	Fatigue	Obtain number of cycles to failure	Bonded titanium lap joint Two gages 0.38 mm (0.015 in.) 1.02 mm (0.040 in.)	Two lap lengths Two temperatures 218°K (-67°F) RT Two stress levels	3	24
	Environmental cycling	Obtain number of environmental exposure cycles to failure[a]	Bonded titanium lap joint Two gages 0.38 mm (0.015 in.) 1.02 mm (0.040 in.)	Two lap lengths	10	20

[a]Environmental cycle: 218°K (-67°F) to 344°K (160°F) at 100% relative humidity; stressed at 30% of the 344°K allowable

TABLE B-3.—Continued

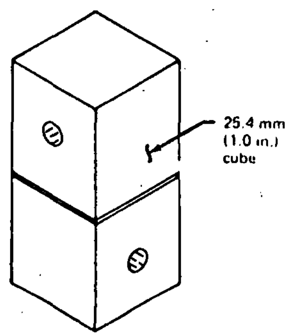
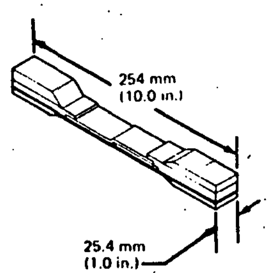
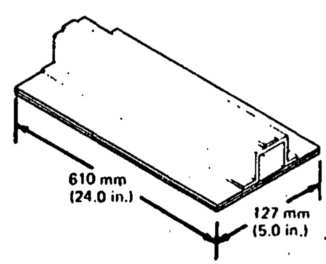
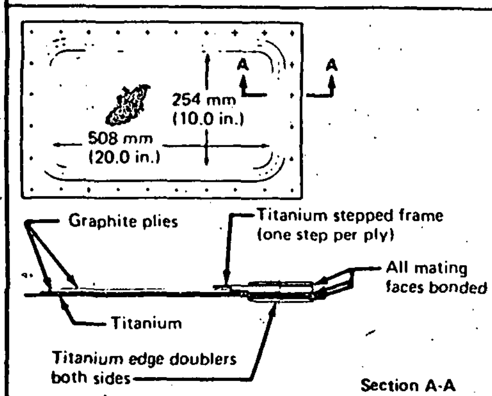
Specimen configuration	Type of test	Test objective	Fabrication details	Variables	Replicates	Total
	Flatwise tension	Obtain ultimate tension load at failure	Bonded titanium blocks Metal-to metal bond	Two adhesive thicknesses	10	20
			Metal-composite-metal bond	Two ply orientations	10	20
	Fatigue of tapered laminates	Obtain number of cycles to failure	Unidirectional 0° plies Bonded titanium fitting on titanium-ends unsymmetric Two gages 0.38 mm (0.015 in.) 0.63 mm (0.025 in.) *Two ply configurations*	Two titanium gages Two ply configurations Two stress levels	3	24
	Environmental cycling	Obtain number of environmental exposure cycles to failure[a]	Unidirectional 0° plies Bonded titanium fitting on titanium-ends unsymmetric Gage 0.63 mm (0.025 in.) Two ply configurations	Two ply configurations	5	10
	Compression ultimate	Obtain ultimate compression load at failure	Unidirectional 0° plies Bonded on stringer Bonded graphite and titanium skin Stringer gages 0.63 mm (0.025 in.) 1.02 mm (0.040 in.) Two laminates (5, 20 plies)	Two metal gages Two ply thicknesses	3	12
	Compression ultimate with void inclusions	Obtain ultimate compression load with *void inclusions*	Unidirectional 0° plies bonded on stringer Bonded graphite and titanium skin Stringer gage 0.63 mm (0.025 in.) Two laminates (5, 20 plies)	Two ply thicknesses *Three void sizes*	3	18

TABLE B-3.—Concluded

Specimen configuration	Type of test	Test objective	Fabrication details	Variables	Replicates	Total
	Shear ultimate	Obtain ultimate shear stress of unstabilized laminate	±45° plies bonded on titanium skin with picture frame edge Two titanium gages 0.38 mm (0.015 in.) 0.51 mm (0.020 in.) Two laminates (2,8 plies)	Two metal gages Two ply thicknesses	5	20

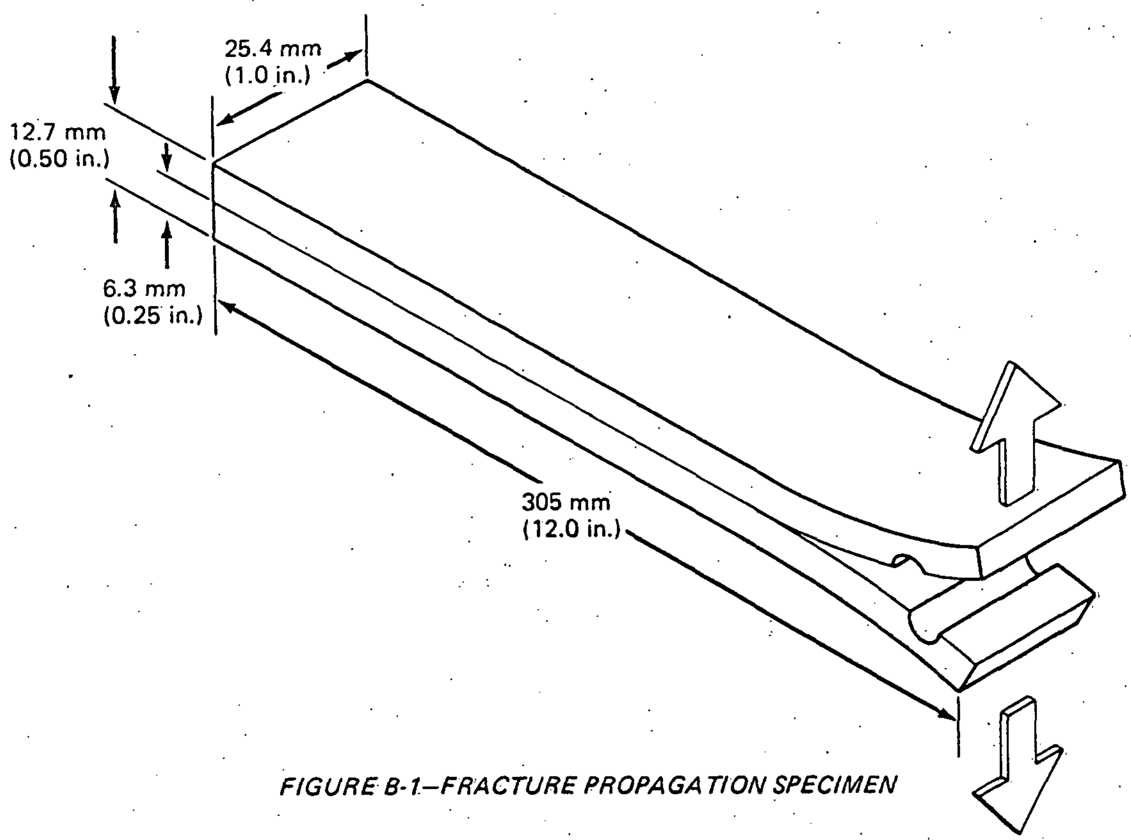


FIGURE B-1—FRACTURE PROPAGATION SPECIMEN

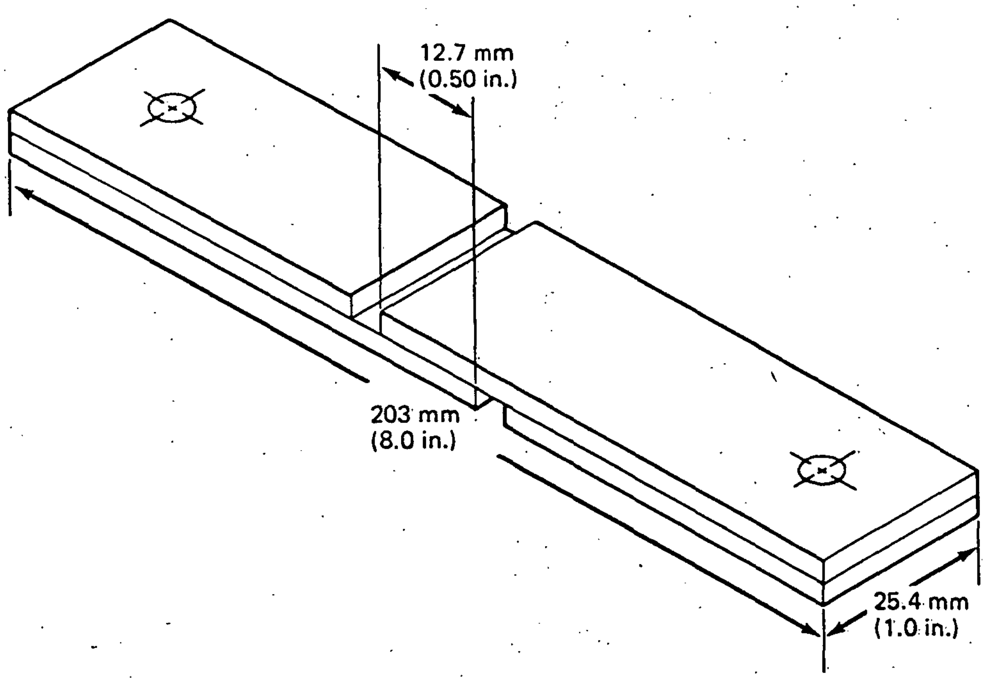


FIGURE B-2—LAP SHEAR SPECIMEN

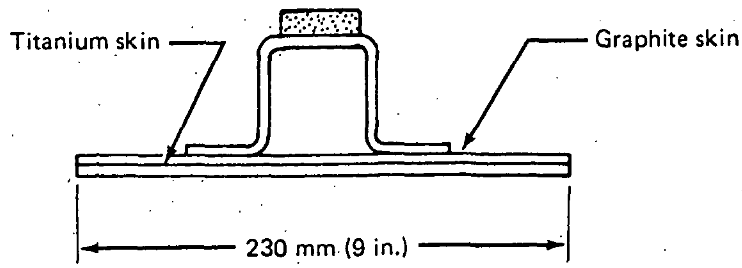


FIGURE B-3.—RESIDUAL THERMAL STRESS FATIGUE TEST SPECIMEN SECTION—CONCEPT 2

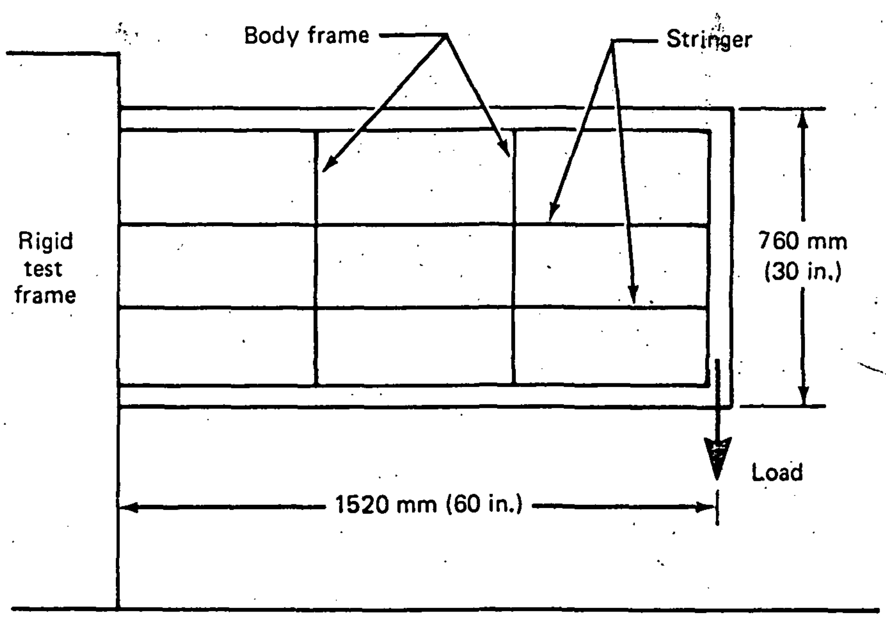


FIGURE B-4.—BONDED SHEAR WEB STIFFENER TEST SPECIMEN CONFIGURATION

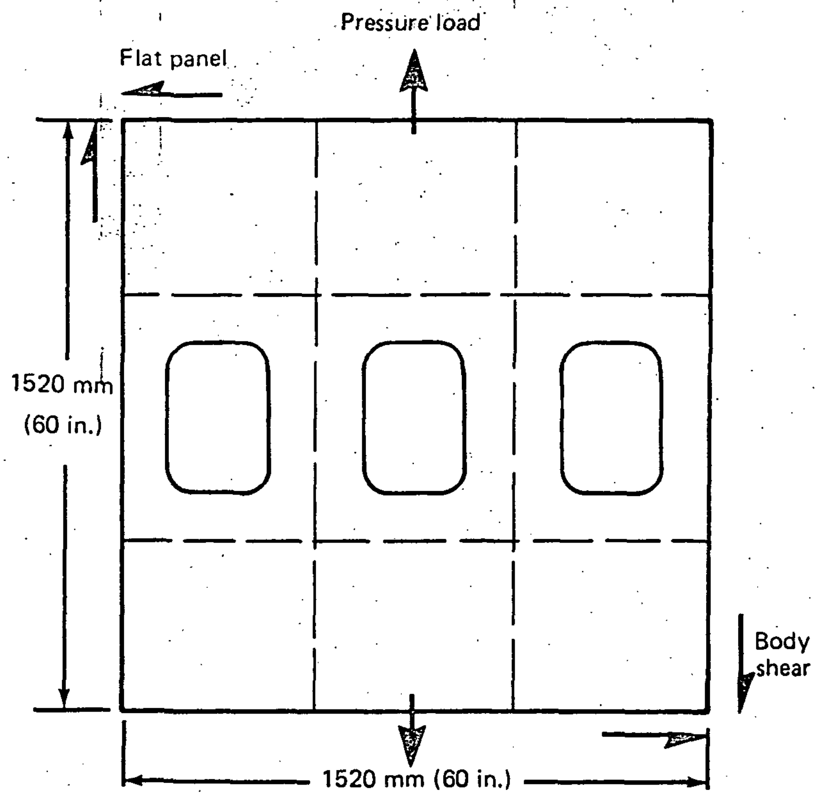


FIGURE B-5.—WINDOW PANEL ULTIMATE AND FATIGUE TEST SPECIMEN CONFIGURATION—CONCEPT 2

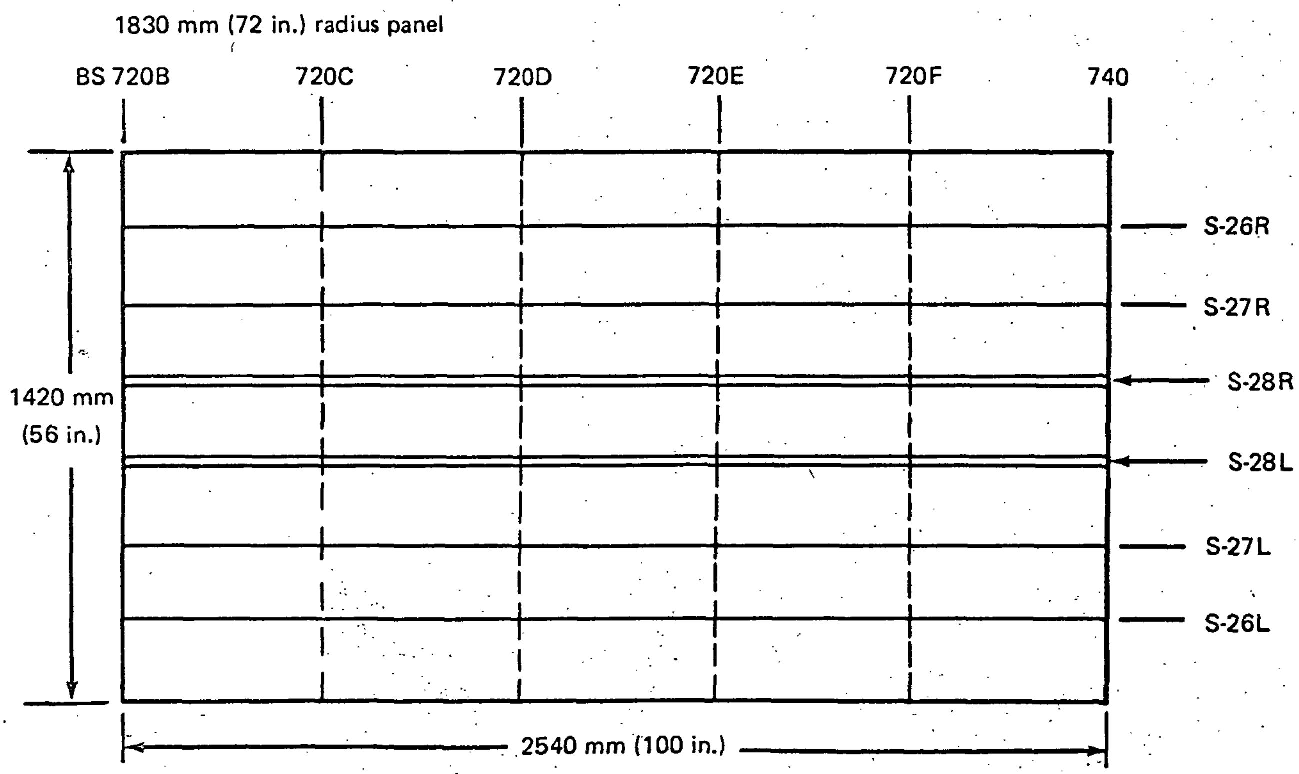


FIGURE B-6.—LOWER QUADRANT COMPRESSION PANEL TEST SPECIMEN CONFIGURATION—CONCEPT 2

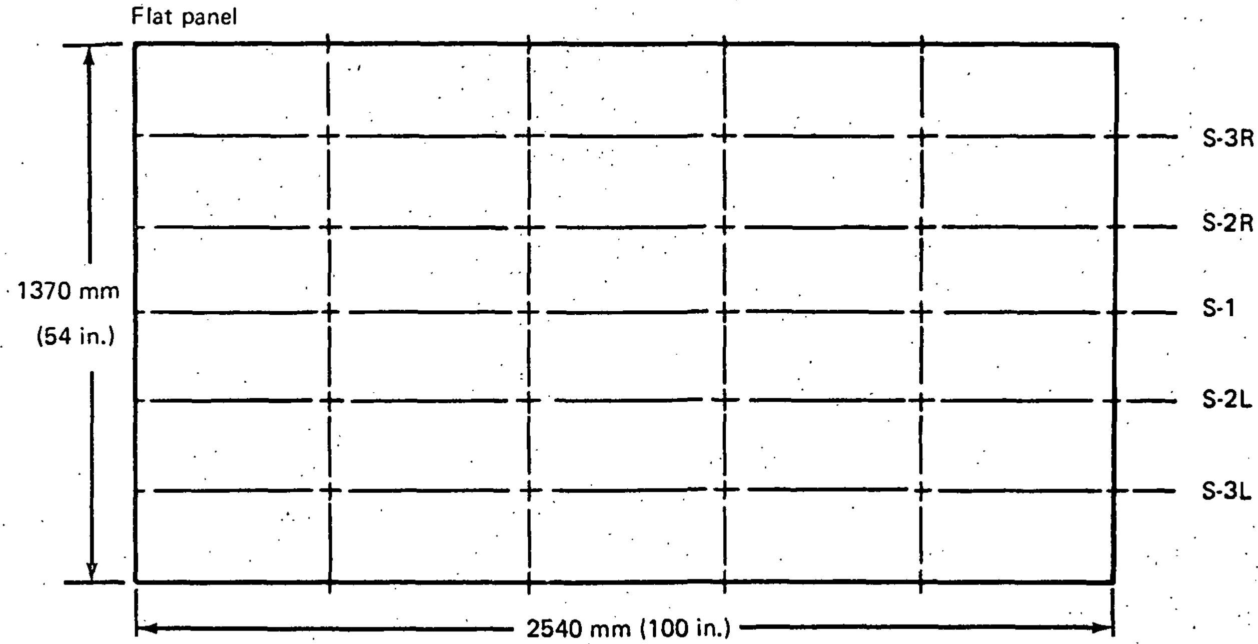


FIGURE B-7.—AXIAL TENSION FAIL-SAFE PANEL TEST SPECIMEN CONFIGURATION— CONCEPT 2

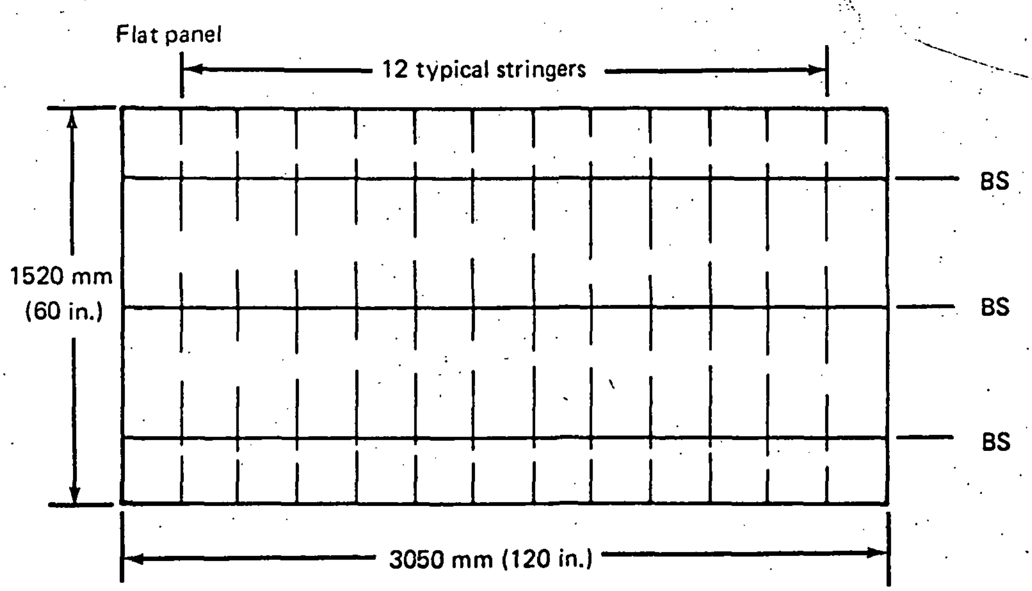


FIGURE B-8.—FRAME TEE FAIL-SAFE STRAP PANEL TEST SPECIMEN CONFIGURATION— CONCEPT 2

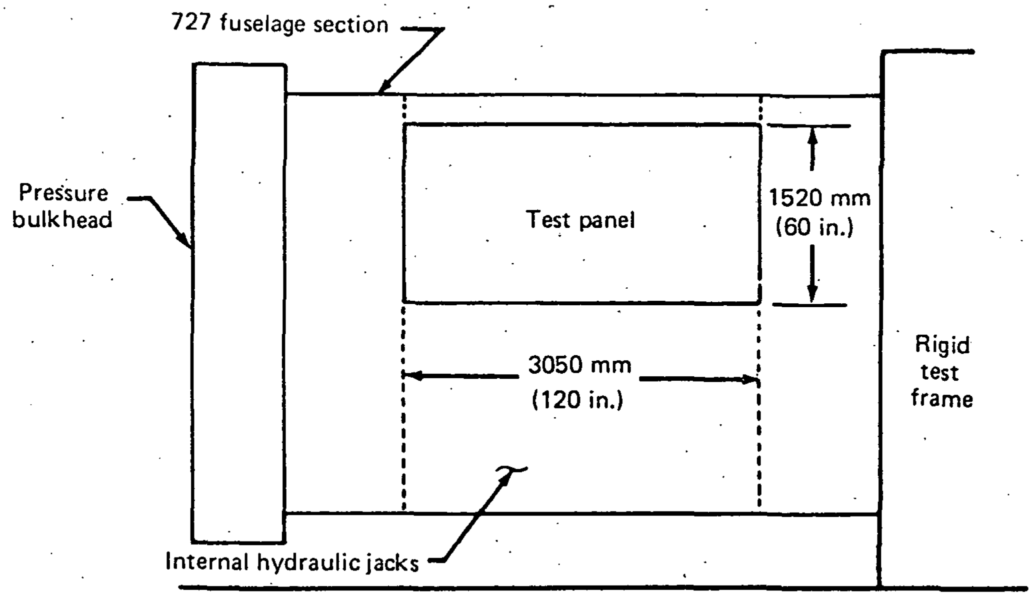


FIGURE B-9.—DAMAGE CONTAINMENT WITH BIAXIAL STRESSES TEST SPECIMEN CONFIGURATION—CONCEPT 2

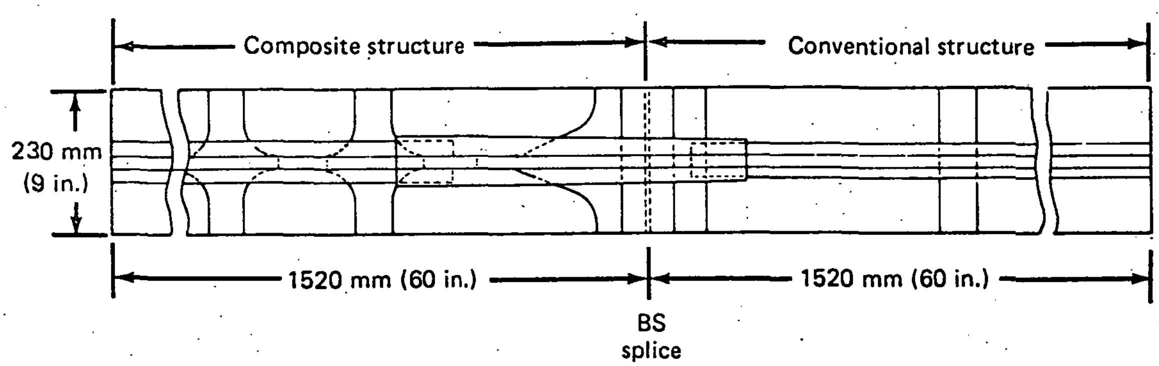


FIGURE B-10.—BODY STATIONS 740 AND 680 CROWN SPLICE FATIGUE TEST SPECIMEN CONFIGURATION—CONCEPT 2

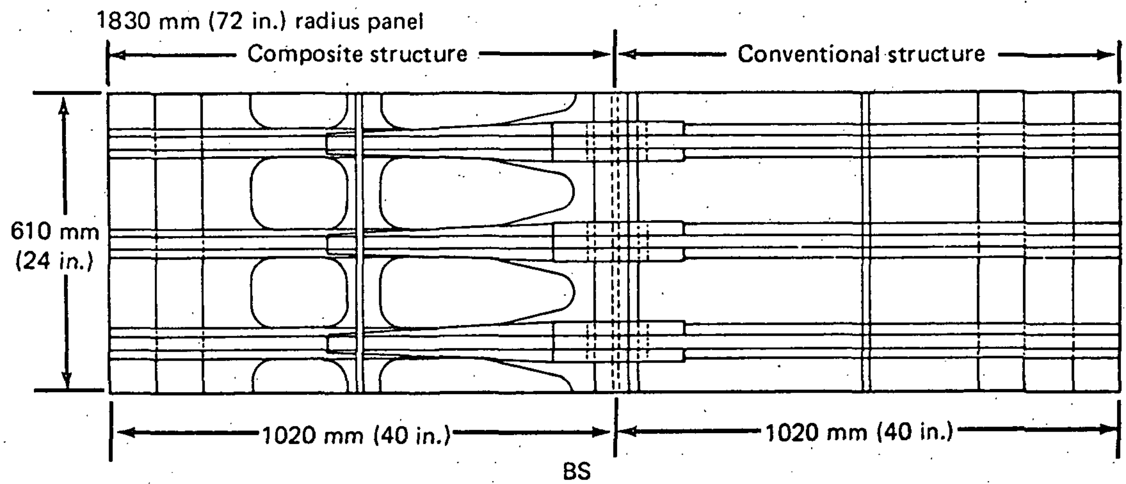


FIGURE B-11.—BODY STATION 680 LOWER QUADRANT SPLICE COMPRESSION TEST SPECIMEN CONFIGURATION—CONCEPT 2

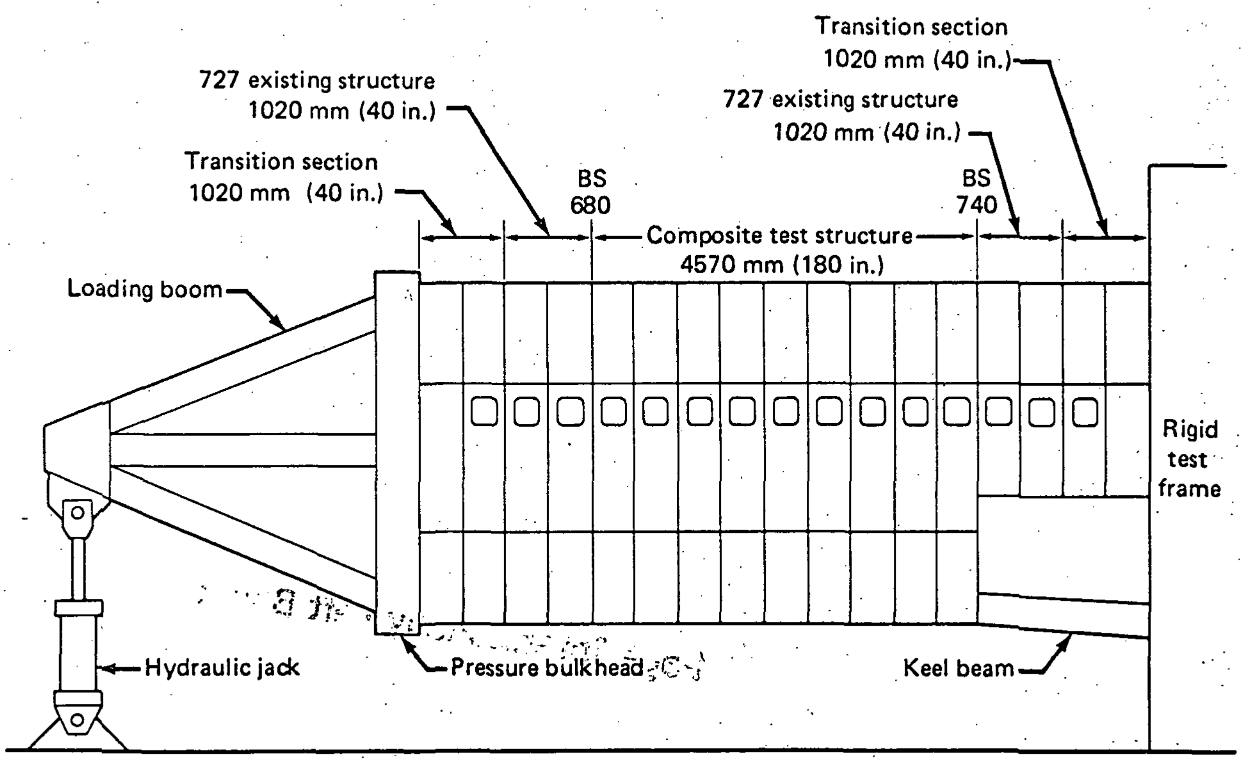


FIGURE B-12.—CONCEPT 2—FULL-SCALE TEST STRUCTURE

APPENDIX C
DETAILED TEST AND MANUFACTURING PLANS FOR THE CONCEPT 3 DEVELOPMENT PROGRAM

C.1 LAMINATE SELECTION TESTS

The composite laminates that were selected during the materials study (sec. 3.3) will be evaluated in the following test program. One composite system will be selected for use throughout the developmental program based on the results of this test program.

Tension, compression, and interlaminar shear tests will be conducted on uniaxial composite at three temperatures, and interlaminar shear tests of bonded titanium and composite will be conducted. In-plane shear tests of ±45° plies will also be conducted. This test program is summarized in table C-1.

C.2 ADHESIVE LABORATORY AND OUTDOOR EXPOSURE TESTS

Laboratory and outdoor exposure tests will be conducted using AF 30 and the selected 394° K (250° F) system. Stressed lap shear and fracture propagation data will be generated. The fracture propagation specimen is shown in figure C-1, the lap shear specimen in figure C-2, and the detailed test plan in table C-2.

C.3 BASIC ALLOWABLES

The basic allowables test program is shown in table C-3.

C.4 MANUFACTURING FEASIBILITY HARDWARE

Full-scale components will be fabricated during this program. These components are defined as follows:

- A 3050-mm (120-in.) long by 2160-mm (85-in.) wide section of the upper quadrant containing the composite skin, the composite reinforced stringer, the frame tees, and the frame J-sections.
- A 3050-mm (120-in.) long by 1520-mm (60-in.) wide section of the window belt panel containing the composite skins, the window frames, and the frame tees.
- A 3050-mm (120-in.) long by 2030-mm (80-in.) wide section of the lower quadrant panel containing the composite skins, the keel beam transition, and the frame tees.

C.5 ENGINEERING VERIFICATION TESTS

The following programs define the engineering verification tests that will be performed for the concept 3 development program.

C.5.1 Test 1—Window Panel Ultimate and Fatigue Tests

This program will evaluate the shear and pressure load capability of a window belt panel (fig. C-3). The window panel will be tested with shear and pressure loads. An ultimate test will be performed on one panel and simulated fatigue cycles will be applied to a second panel. The test panel is a representative section of the window belt area. Panels will be loaded to ultimate pressure and shear loads, so as to obtain strain surveys, ultimate loads, modes of failure, and number of cycles to failure. The instrumentation for each panel will consist of 15 axial gages, 10 crack-wire circuits, 40 rosette gages, and five deflection indicators.

C.5.2 Test 2—Lower Quadrant Compression Panel

This test will evaluate the compression load capability of a representative section of the lower quadrant design (fig. C-4). The panel will be tested by loading the keel beams with concentrated loads and reacting these with a uniform load along the other panel edge. The panel will be curved and the unloaded edges will be simply supported. Two panels will be tested to compression ultimate. The instrumentation on each panel will consist of 40 axial gages and 10 rosette gages.

C.5.3 Test 3—Axial Tension Fail-Safe Panel

This test will evaluate the fatigue crack propagation rates and damage containment capability of a representative upper quadrant panel under axial tension load (fig. C-5). Fatigue cracks will be cut in the panel, the panel will be fatigue cycled for approximately 50 000 cycles, and crack growth rates will be measured. The fatigue cracks will be repaired and blade penetration tests conducted on the panel under axial tension. A total of five panels will be tested with two fatigue crack propagation tests and two blade tests on each panel. The instrumentation on each panel will consist of 40 axial gages and 20 rosette gages.

C.5.4 Test 4—Frame Tee Fail-Safe Strap Panel Test

This test will evaluate the fatigue crack propagation rates and damage containment capability of a representative quadrant panel under pressure load (fig. C-6). Fatigue cracks will be cut in the panel, the panel will be fatigue cycled for approximately 50 000 cycles, and crack growth rates will be measured. The fatigue cracks will be repaired and blade penetration tests conducted on the panel under hoop pressure tension loads. A total of five panels will be tested with two fatigue crack propagation tests and two blade tests on each panel. The instrumentation on each panel will consist of 40 axial gages and 20 rosette gages.

C.5.5 Test 5—Damage Containment with Biaxial Stresses

This test will evaluate the damage containment capability of a fuselage skin panel under pressure stress and axial tension or compression (fig. C-7). A representative section of a skin panel will be installed in a section of a 727 fuselage contained between two pressure bulkheads. The fuselage section will be pressurized and axial tension or compression will be introduced into the shell by hydraulic jacks acting between the end bulkheads. Blade penetration tests will be conducted on the test panel under biaxial stress. Five panels will be tested with four blade shots in each panel. The instrumentation on each panel will consist of 40 axial gages and 20 rosette gages.

C.5.6 Test 6—Body Station 740 and 680 Crown Splice Fatigue Tests

These tests will evaluate the fatigue capability of the BS 740 and 680 skin splice details (fig. C-8). Three test panels that represent each area will be fatigue tested to failure. The instrumentation on each panel will consist of 20 axial gages.

C.5.7 Test 7—Body Station 680 Lower Quadrant Splice Compression Test

This test will evaluate the ultimate compression capability of the BS 680 compression splice (fig. C-9). Three test panels that represent this splice area will be tested to failure. The instrumentation on each panel will consist of 40 axial gages.

C.6 FABRICATION PROCESSES

The fabrication processes for the upper quadrant panel are defined as follows:

- The fiberglass stringers will be formed and cured with the frame tee joggles included.
- The stringer composite reinforcing will be cured and bonded to the step fittings with AF 30 adhesive as a subassembly.
- The stringer composite assembly will be bonded to the stringers with AF 30 adhesive.
- The composite skin will be cured and bonded to the precurved picture frame with AF 30 adhesive on a flat tool. The assembly will then be vacuum bagged and heated to 394° K (250° F) in a curved bond tool.
- The stringers, composite skin, and frame tees will be bonded with the 394° K (250° F) adhesive in a curved bond tool.
- The frame J-sections will be mechanically fastened with conventional processes.

The fabrication processes for the side quadrant panel are defined as follows:

- The composite skins will be cured and bonded to the precurved picture frame and window frame load transfer rings with AF 30 adhesive on a flat tool. The assembly will then be vacuum bagged and heated to 394° K (250° F) in a curved tool.

- The outer skin and core will be bonded with the 394° K (250° F) adhesive in a curved bond tool.

- The skin-core assembly will be pulled flat and the core will be machined to a flat plane.

- The inner skin and window frames will be bonded to the core with the 394° K (250° F) adhesive in a curved bond tool.

- The frame tees will be bonded to the skin assembly with the 394° K (250° F) adhesive.

- The frame J-sections will be mechanically fastened with conventional processes.

The fabrication processes for the lower quadrant panel are defined as follows:

- The composite skins will be cured and bonded to the precurved picture frame with AF 30 adhesive on a flat tool. The assembly will then be vacuum bagged and heated to 394° K (250° F) in a curved tool.

- The outer skin and core will be bonded with the 394° K (250° F) adhesive in a curved bond tool.

- The skin core assembly will be pulled flat and the core will be machined to a flat plane.

- The inner skin and keel beam fitting will be bonded to the skin assembly with the 394° K (250° F) adhesive in a curved bond tool.

- The frame tees will be bonded to the skin assembly with the 394° K (250° F) adhesive.

- The frame J-section and keel beam center skin panel will be mechanically fastened with conventional processes.

- The body-wing fairing panels will be mechanically fastened to fittings bonded on the exterior of the panel.

The fabrication processes for the floor beams are defined as follows:

- The composite chords will be cured as a subassembly.

- The composite, cap strips, web channel sections, doublers, and core will be bonded with the 394° K (250° F) adhesive.

C.7 QUALITY ASSURANCE FOR FABRICATION

During each phase of fabrication, nondestructive inspections will be performed on all components. The fabrication sequences have been arranged so as to provide easy access to all bond lines. Water-coupled through-transmission ultrasonic inspection will be the primary process used for quality assurance control because of its high resolution of flaw detection. The detailed quality control procedures for the concept 3 components are defined as follows:

- The cured stringer and skin-reinforcing laminate and the step fitting bonds will each be inspected as subassemblies.
- The cured composite skins and bonded step fittings will be inspected as a subassembly.
- The bonded sandwich panel, frame tees, and keel beam fitting will be inspected as a final assembly.
- Quality assurance methods for the floor beam assembly will be defined by the developmental program in phase I.
- Quality assurance of the fastener installation and final assembly will be controlled by standard inspection procedures.

C.8 FULL-SCALE GROUND TEST

The full-scale ground test will consist of a complete concept 3 fuselage section 4570-mm (180-in.) long with a 2030-mm (80-in.) long transition section of conventional 727 structure on each end. The test fuselage will be mounted as a cantilever beam and the loads will be introduced through a loading boom. The test setup is shown in figure C-10. Compressed air will be used for fuselage pressurization and styrofoam will be used to reduce the air volume. The static loads will consist of seven conditions of positive and negative bending moment, ground handling, and pressure. The fatigue loading will be a ground-air-ground spectrum consisting of taxi, positive vertical, negative vertical, and pressure loading. Magnetic digital tape programs will be used to control the loads. A computer-controlled digital data acquisition system will be used to record all data. The following instrumentation will be used on this test.

- 150 axial strain gages
- 60 rosette strain gages
- 15 electrical deflection indicators

- 10 crack-wire circuits
- 1.86 sq m (20 sq ft) of photoelastic coating located in 10 separate locations
- 5 inertia-triggered strobe flash camera circuits
- 10 sound detection circuits

During the cyclic fatigue testing, four major inspections will be performed for each lifetime. The test program sequence is as follows.

- Load to 75% limit load, obtain strain survey
- Conduct cyclic fatigue test, four lifetimes
- Conduct seven static load surveys to limit load
- Conduct seven static load conditions to ultimate

TABLE C-1.–LAMINATE SELECTION TEST PROGRAM–CONCEPT 3

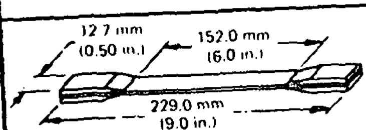
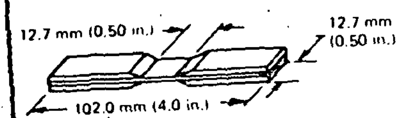
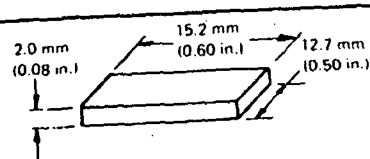
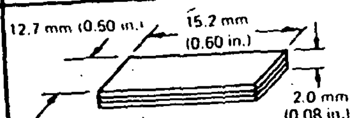
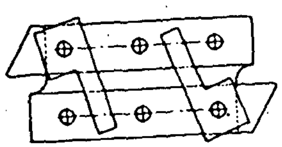
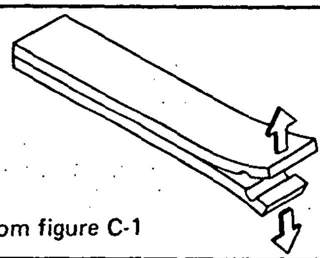
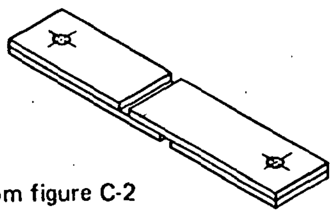
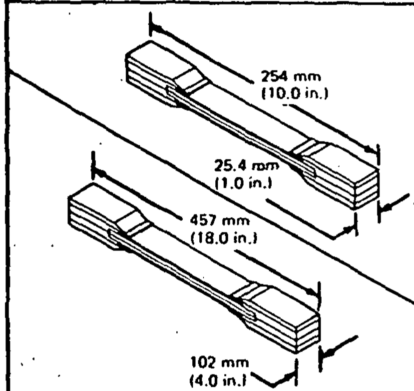
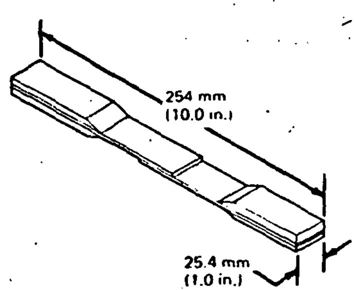
Type of test	Specimen configuration	Test objective	Fabrication details	Test temperature	Composites	Replicates	Total
Tension		Modulus ultimate strength	Unidirectional 0° six plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Compression		Modulus ultimate strength	Unidirectional 0° six plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Interlaminar shear		Shear stress	Unidirectional 0° 16 plies	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
Interlaminar shear composite metal bond		Bond shear	Composite six plies Titanium 0.50 mm (0.020 in.)	RT, 218°K (-67°F), 344°K (160°F)	3	6	54
In-plane shear		In-plane shear strength	±45° laminates four plies	RT	3	6	18

TABLE C-2.–ADHESIVE LABORATORY AND OUTDOOR EXPOSURE TEST PROGRAM–CONCEPT 3

Type of test	Specimen configuration	Adhesives	Adherend sets	Specimens		Stress levels	Total specimens exposed	Control specimens
				Outdoor exposure	Laboratory exposure			
Fracture propagation	 From figure C-1	2	Ti-Ti Comp-comp Ti-comp	4	4	1	48	20
Lap shear	 From figure C-2	2	Ti-Ti Comp-comp Ti-comp	4	4	[a]4	192	50

[a]Stress levels 0%, 10%, 20%, and 30% of the 344° K (160°F) allowable

TABLE C-3.–BASIC ALLOWABLES TEST PROGRAM–CONCEPT 3

Specimen configuration	Type of test	Test objective	Fabrication details	Variable	Replicates	Total
	Tension ultimate	Obtain ultimate tension load at failure	Unidirectional 0° plies Bonded titanium fitting Three laminates (4, 8, 16 plies)	Three thicknesses Three temperatures 218° K (-67°F) RT 344° K (160°F)	3	27
	Fatigue	Obtain number of cycles to failure	Unidirectional 0° plies Bonded titanium fitting Two laminates (4, 16 plies)	Two thicknesses Two temperatures 218° K (-67°F) RT Two stress levels	3	24
	Environmental cycling	Obtain number of environmental exposure cycles to failure[a]	Unidirectional 0° plies Bonded titanium fitting Two laminates (4, 16 plies)	Two thicknesses	10	20
	Fatigue with voids	Obtain number of cycles to failure	Unidirectional 0° plies Bonded titanium fitting with void inclusions Two laminates (4, 8 plies)	Two thicknesses Four void sizes	3	24
	Tension ultimate	Obtain ultimate tension load at failure	Bonded titanium lap joint Three gages 0.38 mm (0.015 in.) 0.51 mm (0.020 in.) 1.02 mm (0.040 in.)	Three lap lengths Three temperatures 218° K (-67°F) RT 344° K (160° F)	3	27
	Fatigue	Obtain number of cycles to failure	Bonded titanium lap joint Two gages 0.38 mm (0.015 in.) 1.02 mm (0.040 in.)	Two lap lengths Two temperatures 218° K (-67°F) RT Two stress levels	3	24
	Environmental cycling	Obtain number of environmental exposure cycles to failure[a]	Bonded titanium lap joint Two gages 0.38 mm (0.015 in.) 1.02 mm (0.040 in.)	Two lap lengths	10	20

[a]Environmental cycle: 218° K (-67° F) to 344° K (160° F) at 100% relative humidity; stressed at 30% of the 344° K allowable.

TABLE C-3.—Continued

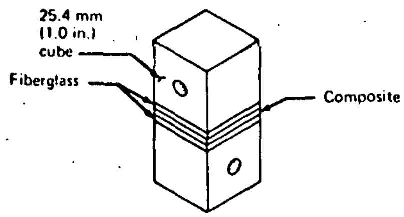
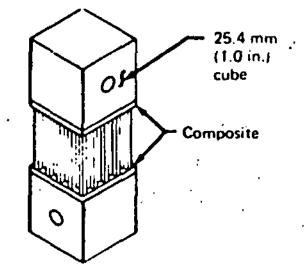
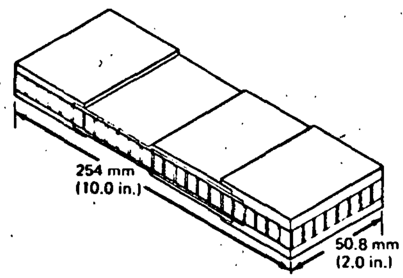
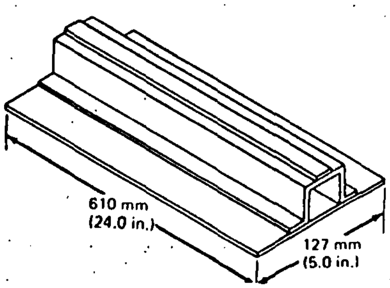
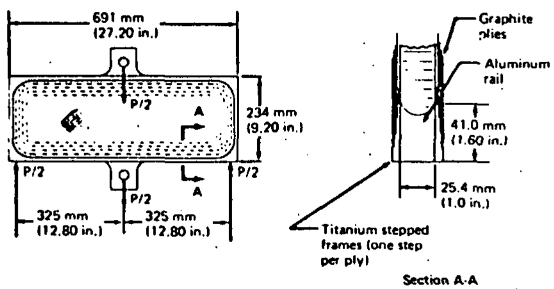
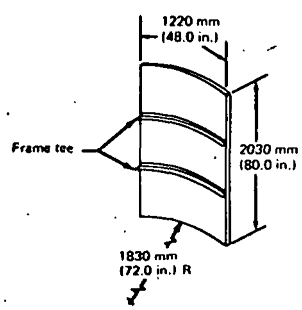
Specimen configuration	Type of test	Test objective	Fabrication details	Variables	Replicates	Total
	Flatwise tension	Obtain ultimate tension load at failure	Bonded fiberglass and composite Fiberglass thickness 0.76 mm (0.030 in.) Composite thickness 0.63 mm (0.025 in.)	Two adhesive thicknesses	10	20
	Flatwise tension	Obtain ultimate tension load at failure	Bonded composite to honeycomb Composite thickness 0.63 mm (0.025 in.)	Two adhesive thicknesses	10	20
	Fatigue of tapered laminates	Obtain number of cycles to failure	Unidirectional 0° plies bonded on honeycomb core Unsymmetric ends	Three ply thicknesses Two stress levels	3	18
	Environmental cycling	Obtain number of environmental exposure cycles to failure[a]	Unidirectional 0° plies bonded on honeycomb core Unsymmetric ends	Two ply thicknesses	5	10
	Compression ultimate	Obtain ultimate compression load at failure	Unidirectional 0° plies bonded on fiberglass stringer Bonded graphite skin Two laminates (5, 15 plies) Stringer gage 0.68 mm (0.027 in.)	Two ply thicknesses	3	6
	Compression ultimate with void inclusions	Obtain ultimate compression load with void inclusions	Unidirectional 0° plies bonded on fiberglass stringer Bonded graphite skin Two laminates (5, 15 plies) Stringer gage 0.68 mm (0.027 in.) Three void sizes	Two ply thicknesses Three void sizes	3	18

TABLE C-3.—Concluded

Specimen configuration	Type of test	Test objective	Fabrication details	Variables	Replicates	Total
	Shear ultimate	Obtain ultimate shear stress of stabilized laminates	±45° plies bonded on honeycomb core with picture frame edge Three ply thicknesses	Three ply thicknesses	5	15
	Compression ultimate	Compression ultimate load of honeycomb stabilized composite skins	Composite laminate skins bonded on honeycomb core with frame tees Two ply thicknesses	Two ply thicknesses	3	6
	Compression ultimate with void inclusions	Compression ultimate load of honeycomb stabilized composite skins with void inclusions	Composite laminate skins bonded on honeycomb core with frame tees and void inclusions Two void sizes Two ply thicknesses	Two void sizes Two ply thicknesses	3	12

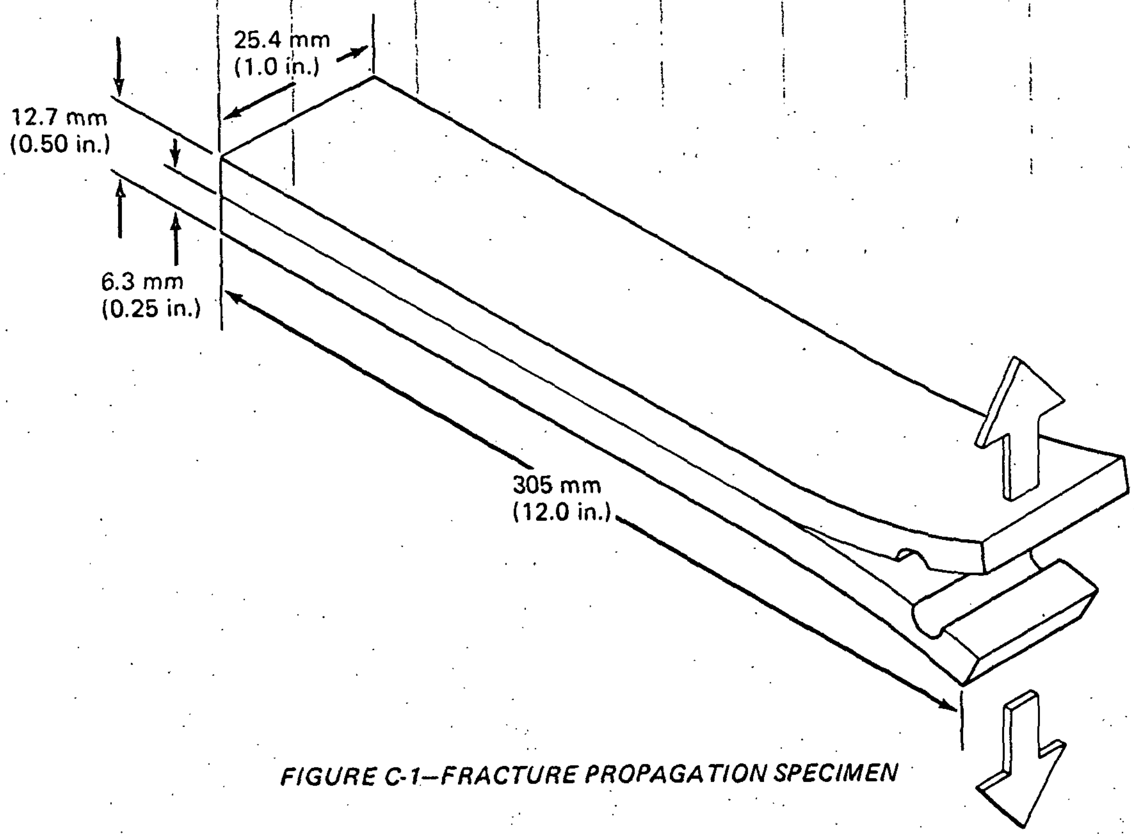


FIGURE C-1—FRACTURE PROPAGATION SPECIMEN

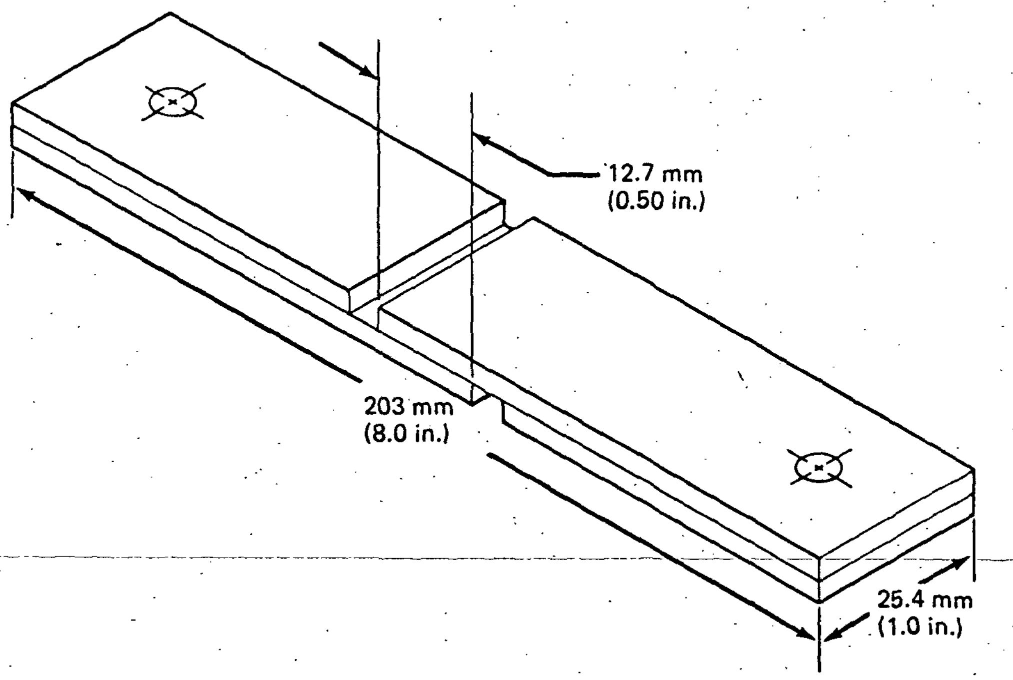


FIGURE C-2—LAP SHEAR SPECIMEN

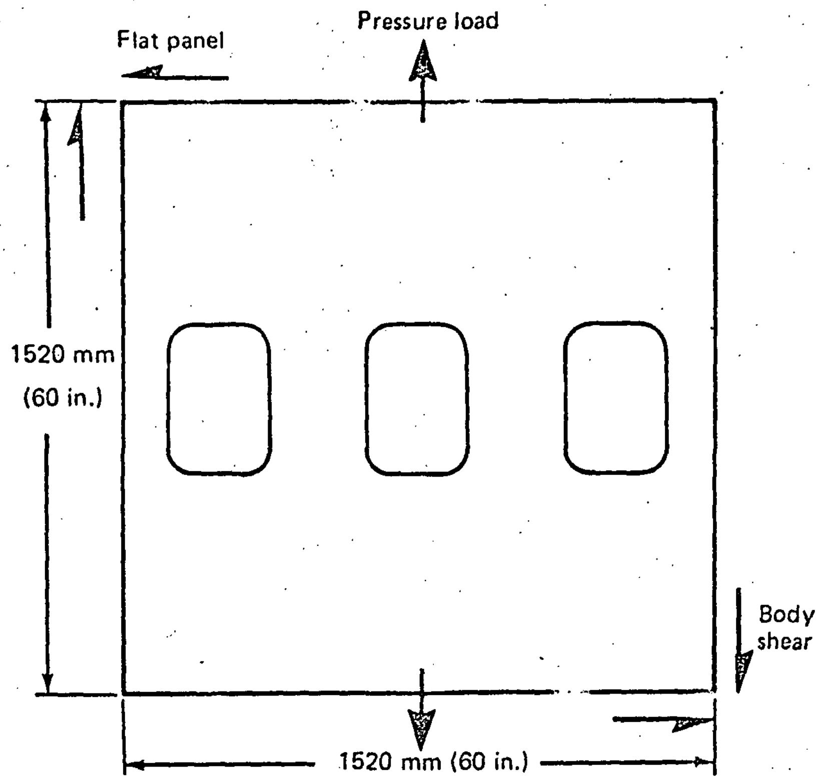


FIGURE C-3.—WINDOW PANEL ULTIMATE AND FATIGUE TEST SPECIMEN CONFIGURATION—CONCEPT 3

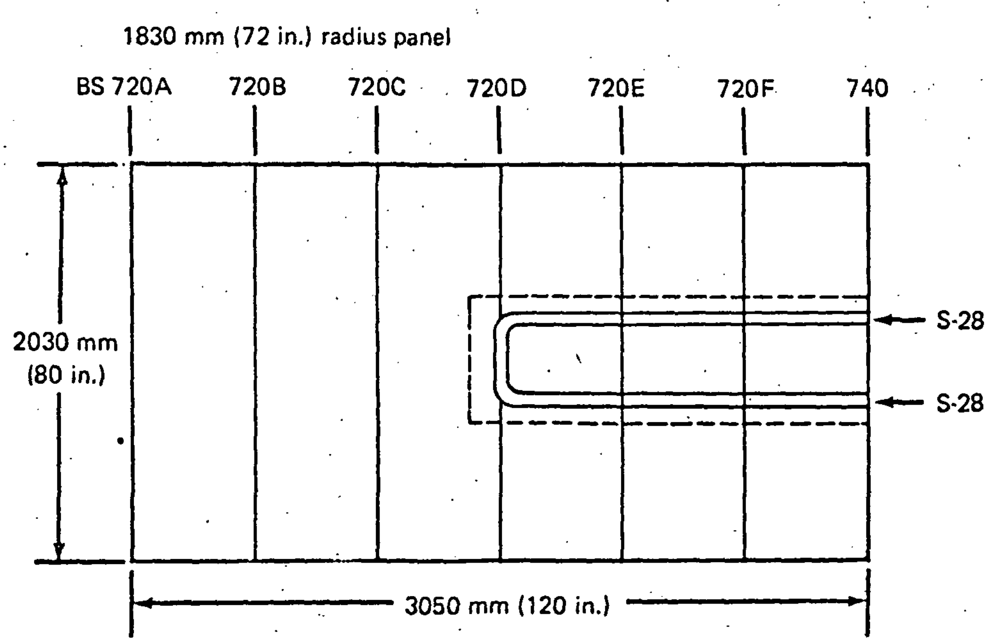


FIGURE C-4.—LOWER QUADRANT COMPRESSION PANEL TEST SPECIMEN CONFIGURATION—CONCEPT 3

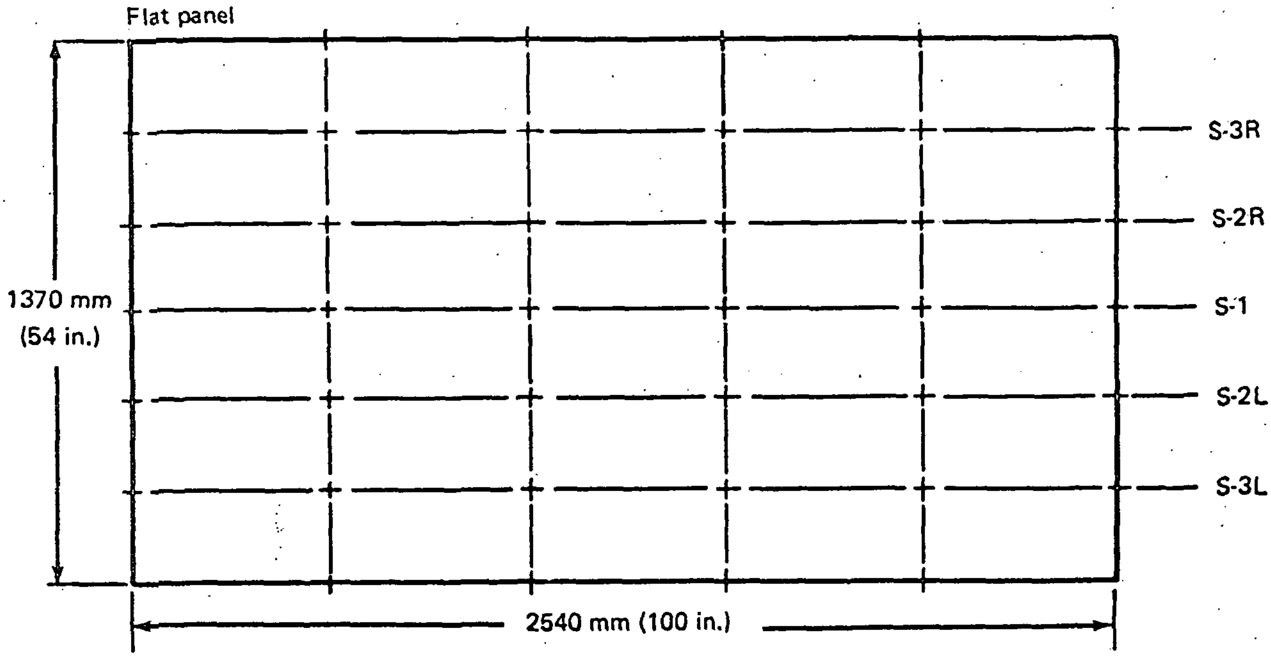


FIGURE C-5.—AXIAL TENSION FAIL-SAFE PANEL TEST SPECIMEN CONFIGURATION— CONCEPT 3

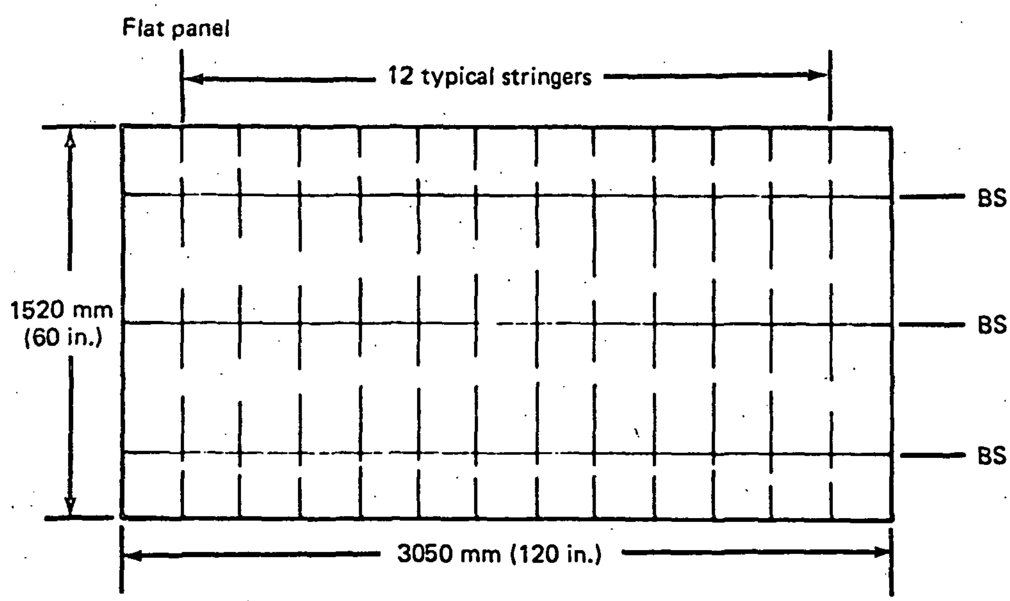


FIGURE C-6.—FRAME TEE FAIL-SAFE STRAP PANEL TEST SPECIMEN CONFIGURATION— CONCEPT 3

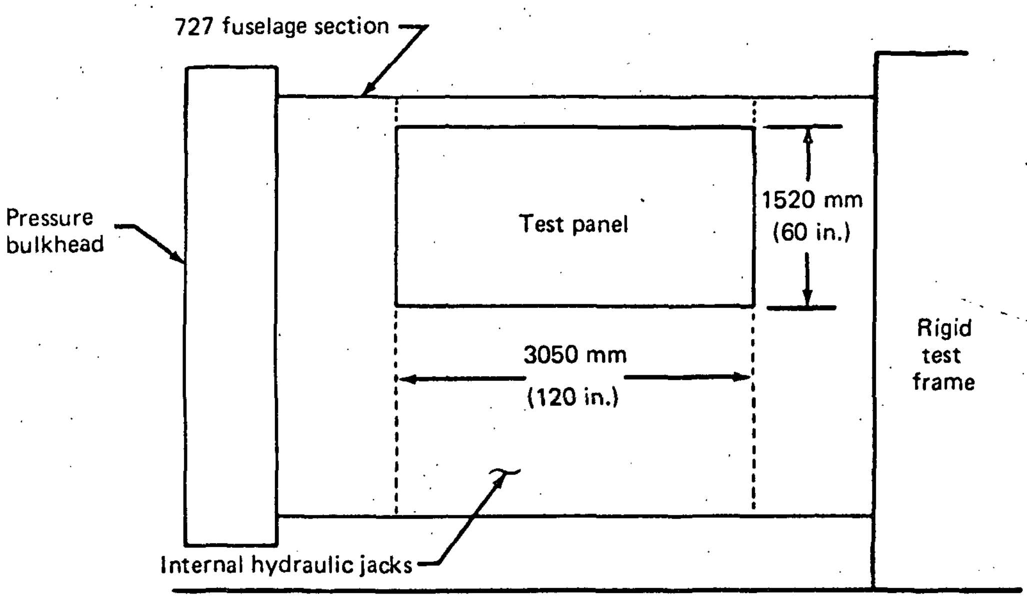


FIGURE C-7.—DAMAGE CONTAINMENT WITH BIAXIAL STRESSES TEST SPECIMEN CONFIGURATION—CONCEPT 3

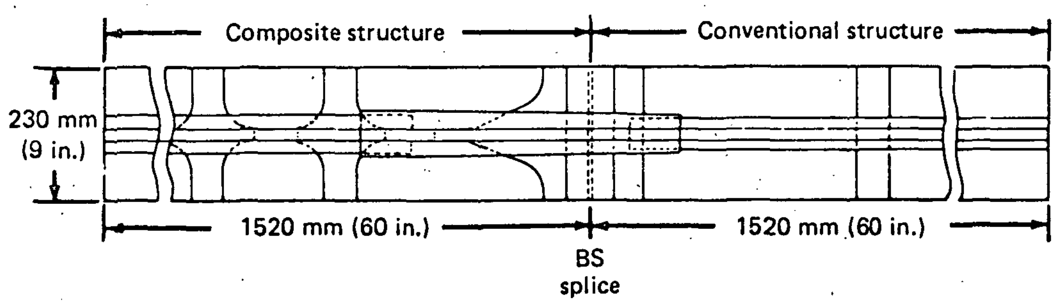


FIGURE C-8.—BODY STATION 740 AND 680 CROWN SPLICE FATIGUE TEST SPECIMEN CONFIGURATION—CONCEPT 3

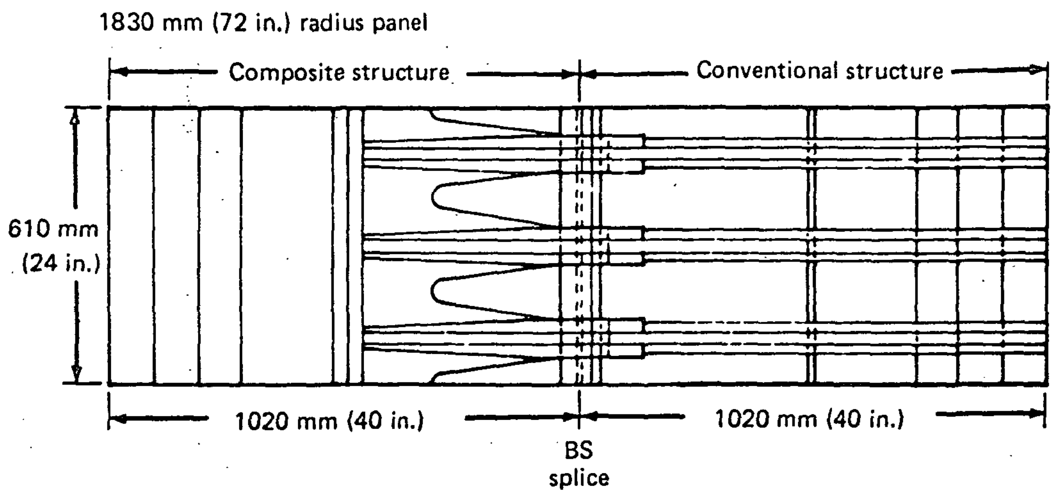


FIGURE C-9.—BODY STATION 680 LOWER QUADRANT SPLICE COMPRESSION TEST SPECIMEN CONFIGURATION—CONCEPT 3

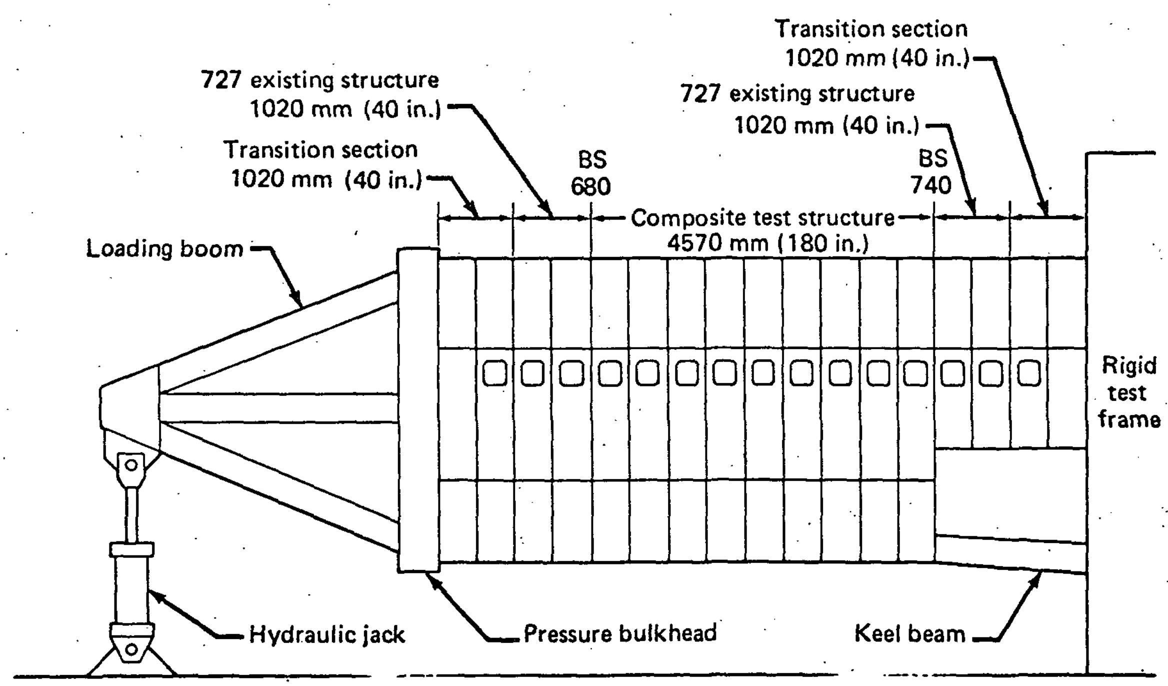


FIGURE C-10.—CONCEPT 3—FULL-SCALE TEST STRUCTURE

APPENDIX D
CONVERSION OF SI UNITS TO U.S. CUSTOMARY UNITS

The international system of units (SI) was adopted by the Eleventh General Conference on Weights and Measures, Paris, October 1960. Conversion factors for the units used herein are given in the following table:

Physical quantity	SI unit	Conversion factor[a]	U.S. customary unit
Length	Meter (m)	39.37	in.
Area	Square centimeter (sq cm)	0.155	sq in.
Mass	Kilogram (kg)	2.2045	lbm
Load	Newton (N)	0.2248	lbf
Density	Gram/cubic centimeter (g/cu cm)	0.0361	lbm/cu in.
Area density	Kilogram/square meter (kg/sq m)	0.2048	lbm/sq ft
Modulus, strength, pressure	Meganewton/square meter (MN/sq m)	145.0	lbf/sq in.
Temperature	^{o}K (Kelvin)	$\frac{9}{5} t_k - 460$	^{o}F (Fahrenheit)

[a]*Multiply the value in SI units by the conversion factor to obtain the value in U.S. customary units.*

Prefixes to indicate multiples of units are as follows:

milli (m)	10^{-3}
centi (c)	10^{-2}
kilo (k)	10^{3}
mega(M)	10^{6}
giga (G)	10^{9}

APPENDIX E
TABLE OF DENSITIES AND WEIGHTS USED FOR WEIGHT-SAVING CALCULATIONS

Material	Density	
	g/cu cm	lbm/cu in.
Graphite composite	1.58	0.057
PRD-49	1.36	0.049
Adhesive	1.22	0.044
Aluminum—2024-T3	2.77	0.100
Aluminum—7075-T6	2.80	0.101
Titanium—6Al-4V	4.44	0.160
Fiberglass	1.94	0.070

Material	Weight	
	kg/sq m	lbm/sq ft
External wire mesh screen for concept 3	0.205	0.042

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3. *Advanced Composites Design Guide,* Air Force Materials Laboratory, third edition, November 1971.

4. *Coatings for Lightning Protection of Structural Reinforced Plastics, Part 1,* Technical Report AFML-TR-70-303, March 1971.

5. *SAMECS Structural Analysis System—User's Document, Versions S0172BH, S0172BF,* D6-23757-1TN, The Boeing Company, February 1970.

6. *Analytical and Experimental Investigation of Aircraft Metal Structures Reinforced With Filamentary Composites, Phase III,* prepared under contract NAS1-8858, October 1971.

7. V. Oeverli and A. V. Viswanathan, *BUCLAP–A Computer Program for Uniaxial Compressive Buckling Loads of Orthotropic Laminated Parts,* NASA CR-111869, 1971.

8. V. Oeverli and A. V. Viswanathan, *BUCLAS–A Computer Program for Uniaxial Compressive Buckling Loads of Orthotropic Laminated Structural Sections,* NASA CR-111870, 1971.

9. V. Oeverli and A. V. Viswanathan, *BUCLASP–A Computer Program for Uniaxial Compressive Buckling Loads of Orthotropic Laminated Stiffened Plates,* NASA CR-111871, 1971.